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**Master's Degree in Aeronautical Engineering**

FINAL PROJECT REPORT

# RADIOISOTOPE THERMOELECTRIC GENERATORS AS THE POWER SYSTEM IN A DEEP SPACE CUBESAT MISSION

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# ABSTRACT

ESA as well as NASA have been carrying out studies for developing missions to Uranus and Neptune, known as the ice giants of our solar system. However, several problems are encountered when facing deep space missions such as the observation of these planets. The main objective and theme of this project is the research of the possibility of the usage of a radio thermal generator (RTG) as the main power system supply to a CubeSat. This small satellite's main mission will be focused on orbiting Neptune's biggest moon, Triton, after being launched by a mothership. The advantages and disadvantages of using RTGs as the power system supply are analysed during this paper. Also, the insertion of the CubeSat from the mothership orbiting Neptune to Triton using an orbit transfer is simulated using a specialized software for better visualization of the mission, and to ease the understanding of the reader as to why the power supply system is such a big issue regarding deep space missions.

**Key words: RTG, mission, CubeSat, power, deep space**

# RESUMEN

Tanto la ESA como la NASA han estado llevando a cabo estudios para el desarrollo de misiones a Urano y Neptuno, conocidos como los gigantes de hielo de nuestro sistema solar. Sin embargo, se pueden encontrar varios problemas y dificultades cuando se afrontan misiones de espacio profundo como la que se menciona en este proyecto. El objetivo y tema principal de este trabajo es la investigación de la posibilidad del uso de un generador termoeléctrico de radioisótopos (RTG) como el sistema principal de generador de potencia para un CubeSat. La misión principal de este pequeño satélite se centraría en orbitar la luna más grande de Neptuno, Tritón, tras ser lanzado por una nave nodriza. Las ventajas y desventajas del uso de RTGs como fuente de energía eléctrica serán analizadas durante este documento. Además, se simulará la inserción del CubeSat desde la nave nodriza orbitando Neptuno a Tritón utilizando un software especializado para una mejor visualización de la misión, y para facilitar el entendimiento del lector sobre el porqué el sistema de potencia en misiones de espacio profundo es un problema a tener en cuenta.

**Palabras clave: RTG, misión, CubeSat, potencia, espacio profundo**

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# Chapter 1. INTRODUCTION

## 1.1 Problem approach or statement

Uranus and Neptune, also called the ice giants due to their iced surfaces, have been main targets for interplanetary exploration and scientific and research missions for both the ESA and the NASA.

However, the European association as well as the American, despite working together, have encountered various issues when it comes to deep space missions like the ones that would occur when travelling to these planets. The long distance to the Earth from the space vehicle or satellite would mean having issues as slower communications or less efficient power supply.

This project focuses on the power supply problem faced, discusses the various possibilities and puts the emphasis on the Radioisotope Thermoelectric Generators (RTGs) as a possible solution, backing it up with a research of expected performances.

## 1.2 Project objectives and scope

As previously mentioned, this paper will be focused on not only pointing out the power system issue that these type of long-term missions may face, but will specify the various possibilities to solve this problem. Several solutions will be mentioned generally, and afterwards the aim will be diverted to the RTGs and all the research that has been made about them.

The communications problem, the orbit possibilities and calculations and any other issues are out of the scope of this project. The objective is not to assure that the RTGs are the best option, but to put out there the option of using them experimentally and observe their performance in outer space and longer missions.

So the project objectives can be listed as follows:

- Research the applicability of an RTG to a CubeSat
- Orbit simulation in GMAT software.

## 1.3 Report structure

The structure of this document can be seen in page 7, which shows an overview of the contents and the whole approach and research overtaken throughout it. In each chapter, it will be specified the details of each domain, providing a crossed reference when mentioning data or information from other sections.

A note has to be made regarding the reference for Radioisotope Thermoelectric Generators, which will generally be referred as RTGs most of the parts throughout the paper to ease the reading of the document.

# Chapter 2. MISSION DESCRIPTION

## 2.1 Deep space observation mission

Prior to describing and analysing the CubeSat at hand and previously designed, it is considered important to describe deep space missions in general, and the mission that is wished to be fulfilled by this satellite.

### 2.1.1 Deep space missions

Deep space exploration, a field that includes astronomy, astronautics, and space technology, is dedicated to exploring the remote regions of outer space. However, there is no universally specific definition for what constitutes "remote" regions. In some contexts, it refers to interstellar space. The International Telecommunication Union defines "deep space" as beginning at a distance of 2 million km (approximately 0.01 AU) from Earth's surface, while NASA's Deep Space Network has used criteria ranging from 16,000 to 32,000 km from Earth. This exploration takes place through both human spaceflights (deep-space astronautics) and robotic spacecraft. (NASA, 2015)

Up to now, Voyager 1 holds the record as the farthest space probe launched from Earth, reaching the farther edge of the Solar System on December 5, 2011, and entering interstellar space on August 25, 2012. Beyond the capabilities of this spacecraft, current propulsion technology limitations prevent further deep space exploration. (NASA, 2015)

*Figure 1. Voyager 1. (Actualidad Aeroespacial, 2017)*

Various deep space missions using probes have been undertaken during the last decades, such as both Voyager 1 and 2 missions, Pioneer 10 and 11 or the Cassini-Huygens. None of this missions were structured as CubeSats, nonetheless, all of them faced the same issues regarding these type of missions: orbital mechanics, communications and power supply system.

For this project, a CubeSat was chosen due to the easiness to be deployed from a mothership, as well as the budget being taken into account for possible project profitability purposes, as the aerospace industry relies on investors and their confidence on project developments and success. However, choosing such a small probe, hinders the possible components and their locations inside the structure.

#### 2.1.2 CubeSat mission

In this scenario, the aim is to examine the surface and visual attributes of Triton, Neptune's largest moon. To achieve this objective, a study was conducted to explore various potential tools capable of fulfilling this task. Ultimately, an imaging camera was chosen for the mission.

The pursuit of this scientific goal entails a range of prerequisites that the spacecraft's design must meet. Consequently, these requirements have a direct influence on how the payload and components are integrated and, subsequently, impact the total mass of the satellite. These considerations become crucial in the context of all the calculations belonging to orbital mechanics.

In this case, certain prerequisites require precise spacecraft pointing and orientation to acquire important scientific data. Additionally, a reliable and efficient communications and data handling system is essential to maximize accuracy and utility. As the payload and antennas must be directed towards specific directions, this, in turn, influences the allocation of the other spacecraft components.

The Cubesat in question will be placed in orbit around Triton, relying on a mothership that will, in turn, orbit around Neptune to transmit the collected data. Consequently, specific periods of visibility will occur between them, directly impacting the communications subsystem and overall functionality of the Cubesat.

### 2.2 CubeSat description

The CubeSat as a whole, including its subsystems and design, are briefly shown in this subchapter, as they were previously discussed beforehand in another paper. (Core López, 2022)

#### 2.2.1 Payload

For this mission, the selection of the payload became a decision between an imaging camera and a spectrometer, as they were considered the most intriguing options for data transmission. Eventually, the imaging camera was chosen due to its better suitability for accommodation within the constraints of the small probe and its compact design.

Considering the available cameras in the market and taking into account the size of the Cubesat (16U), it was feasible to choose for a slightly larger camera compared to the ones typically used in geocentric cubesats. This decision allowed for potential betterment in performance and image quality of the captured data.

*Figure 2. SCS Gecko imager. (Satsearch, 2023)*

### 2.2.2 Propulsion subsystem

The typical functions that this subsystem has when referring to the spacecraft at hand, are mainly three:

- Final orbit acquisition (if needed).
- Station-keeping and orbit control.
- Attitude control.

Currently, there are different types of propulsion systems available for spacecrafts, but in this case, two specific ones were discussed: cold gas propulsion and monopropellant hydrazine systems.

Conclusively, the decision was made to choose a cold gas propulsion system due to several reasons:

- Easiness of installation: The cold gas system offers straightforward and efficient installation procedures.
- Low cost: Implementing a cold gas system is relatively economical compared to other propulsion options.

- Low power requirement: Cold gas propulsion demands minimal power, making it energy-efficient.
- Operational simplicity: The system's operational procedures are straightforward and user-friendly.
- Optimization of space inside the spacecraft: The cold gas system allows for efficient utilization of available space within the spacecraft.
- Reduction of the whole system mass: This propulsion choice leads to less propellant usage and simpler hardware, resulting in reduced wet and dry mass of the spacecraft.

The satellite at hand is three-axis stabilized, so the manoeuvres are done about each of the spacecraft's axis. This means that the correction of the orbit and pointing accuracy is to be done on each axis, showcasing the necessity of having canted thrusters.

### 2.2.3 Attitude determination and control subsystem (ADCS)

The Attitude Control Subsystem (ACS) is responsible for maintaining control over the spacecraft's body axes, including pitch, roll, and yaw angles, ensuring they remain within specified limits. This control is crucial for all space vehicles because it ensures that solar panels are consistently oriented towards the sun to meet power requirements. Additionally, considering the selected payload for scientific purposes, the imaging camera must be precisely pointed in a specific direction to capture the desired data effectively. The ACS plays a pivotal role in achieving accurate and successful data retrieval from the imaging camera while also ensuring the spacecraft's overall stability and functionality.

In this case, a hybrid between three-axis and spin stabilization is chosen, being 3-axis stabilized at all times except when hibernation period (between mothership communications).

The decision to opt for spin stabilization over three-axis stabilization is primarily motivated by power considerations. Three-axis stabilization demands significant power, particularly for the thrusters used in controlling the spacecraft's orientation. On the other hand, spin stabilization enables sun tracking simply by relying on the solar panels. This hybrid choice strikes a balance that saves and optimizes power usage, ensuring efficient operation of the spacecraft while meeting the crucial sun-tracking requirements for solar panel orientation.

For this mission, the inclusion of sun sensors is crucial, considering the existence of solar panels on the satellite. Maintaining optimal sun pointing is vital to ensure continuous power generation from the solar panels. Additionally, star trackers play a significant role, particularly in smaller satellites, as they provide valuable information about the spacecraft's relative position and orientation with respect to the stars.

Regarding the potential usage of gyroscopes, there is a possibility of their integration in the spacecraft's instrumentation. Gyroscopes can detect rotation about the axis normal to the orbit plane, and they are often packaged together with other electronics to optimize space utilization.

### 2.2.4 Thermal control subsystem

Triton is $4.488 \cdot 10^{12}$ km from the Sun, so that area of the Solar System is the coolest. Hence, the main issue encountered is about the low temperature faced during the mission.

When designing and selecting thermal systems, the initial step involves incorporating a multilayer insulation (MLI). It is the preferred choice for small satellites due to its attributes, such as cost-effectiveness and compatibility with the limitations in power, mass and volume when it comes to cubesats. MLI guarantees efficient thermal control whilst meeting the specific accommodation requirements for this CubeSat.

The most advantageous approach identified for this subsystem is to integrate body-mounted radiators or heaters, in addition to the thermal insulation layer. This mixture allows for efficient temperature regulation and control within the CubeSat. Moreover, the inclusion of temperature sensors placed strategically throughout the satellite, helps monitor and ensure optimal thermal conditions. By harnessing this system of radiators, heaters, and sensors, the spacecraft can effectively manage its internal temperatures and maintain the required thermal environment for its various components and instruments.

### 2.2.5 On-board data handling subsystem (OBDH)

This subsystem, which can also be called command and data handling (CD&H), is the one responsible of providing commands and managing the following data:

- Science and/or payload data
- Engineering data, which refers to the spacecraft system state and performance

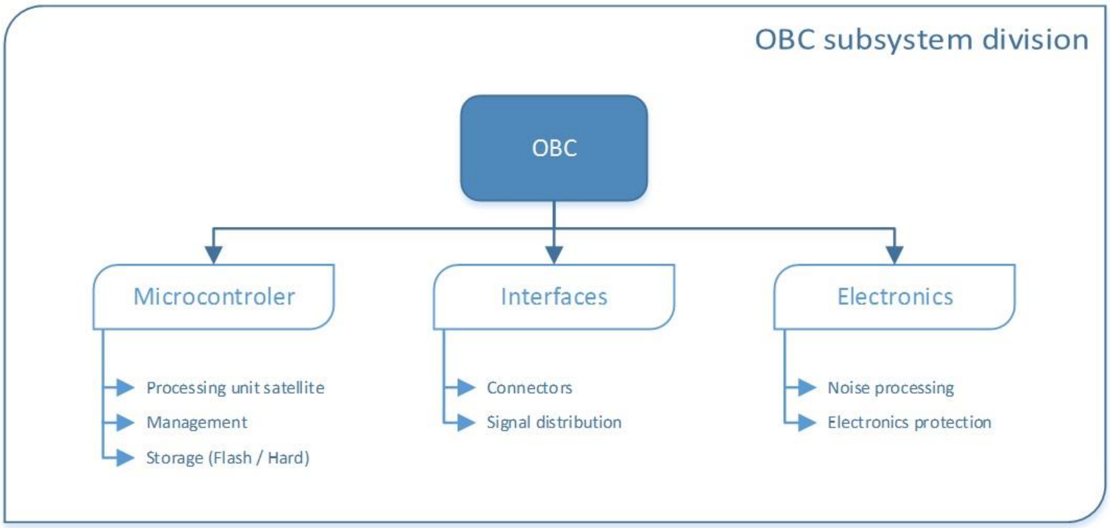



*Figure 3. On-board computer processing block diagram. (ece3sat, 2018).*

For this subsystem, an integrated board is the best option, which would include a micro-processor.

### 2.2.6 Power subsystem

The power subsystem holds a position of utmost importance among all the spacecraft subsystems. Its criticality stems from the fact that a failure in the power supply would result in the mission's overall failure and the loss of the entire space vehicle.

This subsystem is the main research objective in this project, so the explanation of its possibilities and its importance is discussed in further depth in Chapter 4. However, solar arrays are a given in this design as it is one of the specific components that any space mission must have. The calculations for the dimensions of these components are further explained and analysed in Chapter 4.

### 2.2.7 Communications subsystem

In this case, this subsystem withholds the title of one of the most difficult to face issues encountered during deep space missions, as at such a distance from Earth, the communications would be slower and the data would take longer to be transmitted and received by the mission crew on our planet. Furthermore, this subsystem also holds the telemetry, tracking and control system.

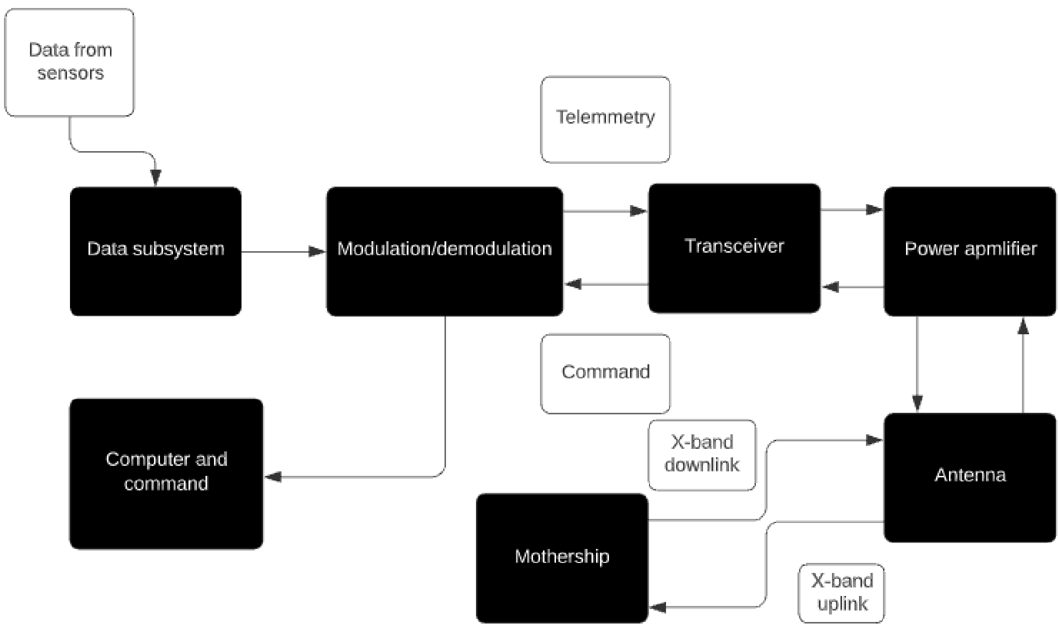



*Figure 4. CubeSat communications system block diagram. (Elaboración propia, 2021)*

Despite the problems that this subsystem might face, it is important to note that the communications with Earth would be transmitted using the mothership orbiting around Neptune, so the CubeSat would be communicating with this mothership, not directly with Earth. This means that the choice for the CubeSat cannot imply inputting a big antenna as in other deep space missions such as the Voyager, as it would be inefficient and physically very difficult to deploy from inside the CubeSat.

Both patched antennas and a small deployable one are interesting choices, besides being realistic and possible, as the main characteristic that they should have for the mission is the pointing to the mothership as many times as possible.

The choice for the communication subsystem is discussed and researched throughout a previous work regarding this mission.

### 2.2.8 Structure and mechanisms subsystem

This subsystem holds significance due to the fact that this type of satellites are needed to be lightweight and mainly low cost, making both characteristics a constraint when designing the CubeSat. As this probe would face extreme temperatures, the best choice of material would be aluminium, which is able to withstand the freezing temperatures that are given at such a distance from the Sun. Moreover, this material complies with the constraints discussed, as it is both light and low cost when compared to other possible choices.

Concerning the solar arrays deployment mechanism, it is important to consider that when they are deployed, it will produce a momentum on the Cubesat. Consequently, the attitude actuators need to be highly accurate and precise to handle this momentum effectively. Furthermore, the ability to store and manage this momentum using the reaction wheels is of crucial importance to maintain the Cubesat's stability and desired orientation throughout the mission.

This Cubesat adheres to the 16U standards, with precise dimensions of 20 x 20 x 40 cm. Each cube has a weight of 1.33 kg, but an ideal structural factor of approximately 15% is applied to enhance robustness. In this report, conservative estimates are taken into account, thus limiting the structural factor to a maximum of 20% of the total anticipated mass for this subsystem.

### 2.2.9 Overall design and components

After choosing the considered best options for the CubeSat, taking into account monetary budget expected as well as the objective of the mission, an approximate mass budget is obtained following the mission requirements approach:

| **Subsystem** | **Specific components** | **Estimated mass (no contingency) [kg]** | **Contingency (%)** |
|---|---|---|---|
| Payload | Gecko imager | 0.480 | 0 |
| Structure & mechanisms | 16 U structure | 4.256 | 15 |
| Thermal control | Temperature sensors (2)<br>Discrete radiators (2)<br>MLI | 0.8 | 15 |
| Power | Solar panels<br>Li-ion battery (2) | 0.5376 + 0.5 x 2 = 1.5376 | 30 |
| TT&C | Patch antenna<br>Transceiver | 1.172 | 30 |
| On-board processing | Micro-processor | 0.1 | 10 |
| Attitude determination and control | Sun sensors (3)<br>Star trackers (3)<br>Reaction wheels (3) | 0.895 | 10 |
| Propulsion | Canted thrusters (4)<br>Propellant | ~ 2 | 30 |
| TOTAL (dry) | - | 9.24 | - |

| | | | |
|---|---|---|---|
| TOTAL (wet) | - | 11.24 | - |

*Table 1. Mass values obtained from mission requirements and analysis. (Core López, 2022)*

Once all the components, the structure and the mass have been analysed and discussed, two possible configurations are shown to better visualize the proposed CubeSat for the mission and for a better understanding of its operability and design, both rendered using the software 360 Fusion (Autodesk) for easiness of outer design.

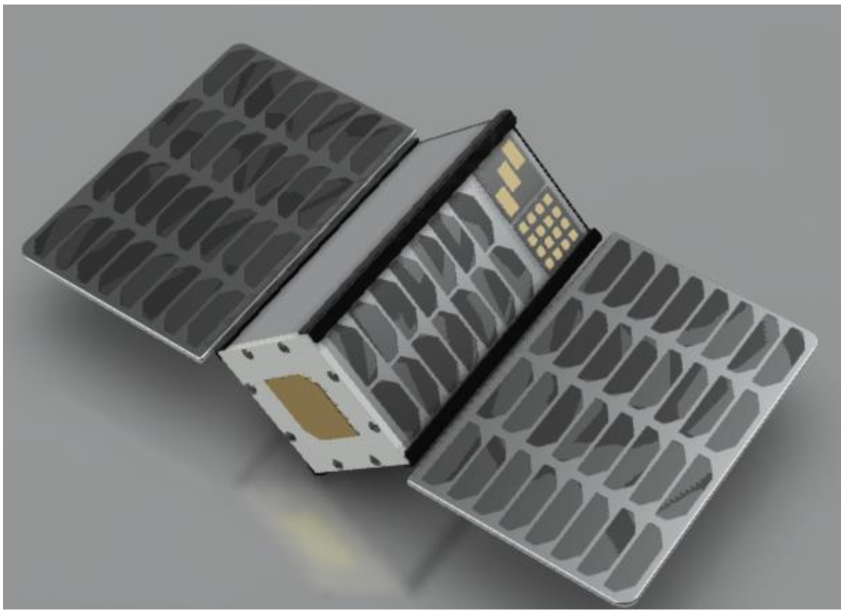


*Figure 5. Main proposed design for the CubeSat. (Elaboración propia, 2021)*

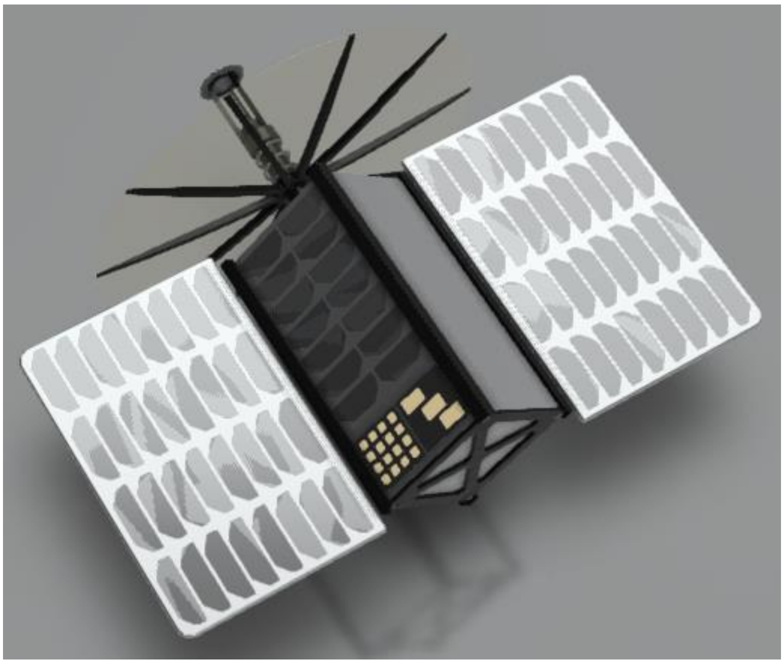


*Figure 6. Alternative proposed design for the CubeSat. (Elaboración propia, 2021)*

In both models, the intended payload is positioned on the free side of the Cubesat, which is the side without any body-mounted solar panel. The precise dimensions of the satellite are provided in section 2.2.8 of the documentation.

# Chapter 3. PROBLEMS ENCOUNTERED

Interstellar and space travel have constantly faced ordeals arising from the immense distances they cross. These challenges includes powering the spacecraft with limited solar array efficiency, maintaining suitable temperature levels amidst the harsh cold environment, and enhancing the effectiveness of long-range communication, among other factors.

This project focuses its objective on proposing a possible solution to the power system problem faced during these type of missions. However, other two problems are briefly mentioned and described, to help the reader understand the difficulties confronted regarding deep space travel and missions by the space industry.

## 3.1 Communications problem

The distance between Earth and Triton amounts to 4338 · 106 kilometres. However, there have been other satellite missions that surpassed this distance and ventured into interstellar space. Voyager 1 and 2 missions achieved interstellar space exploration beyond the Solar System in 2012 and 2018, respectively, while continuously transmitting data to Earth's Deep Space Network (DSN). Additionally, the Cassini-Huygens probe successfully gathered and transmitted essential data from Saturn before its ultimate entry and disintegration into the planet in 2017.

Both of these missions used high gain antennas, however the total mass of these satellites weighted over 700 kilograms each. Therefore, when designing a CubeSat, mass budget is one of the most important constraints faced by engineers, as such small spacecrafts are not able to withstand such huge instruments and components. Despite the CubeSat not communicating directly with Earth but with the mothership instead, using high gain antennas for a CubeSat is both inefficient, expensive and would disrupt the mass budget desired for this project.

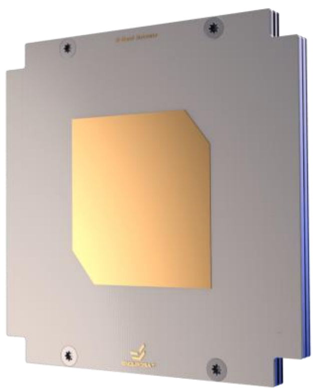


*Figure 7. Characteristic CubeSat patch antenna. (Endurosat, 2021)*

Consequently, the choice for this design was discussed in a previous work, finally being the patched antennas, making it easier to face the mothership when encountering it during orbit. A deployable small antenna is also possible, but it is deemed less reliable and both the structure and mechanisms of the whole design would be affected, proposing this second option as a more innovative, alternative and experimental option.

## 3.2 Orbit problem

Elaborating an orbit is undeniably one of the critical calculations during space mission development, especially for deep space missions.

This mission focuses around Neptune's biggest moon, Triton. But this moon withholds one of the most special and difficult characteristics when facing orbit planning and calculations: its orbit is retrograde. It means that the rotation of its home planet, Neptune, and the rotation of Triton have opposite directions. Furthermore, this moon is similar to our own Earth's moon, as only one of its sides faces Neptune constantly, being locked in a synchronous rotation.

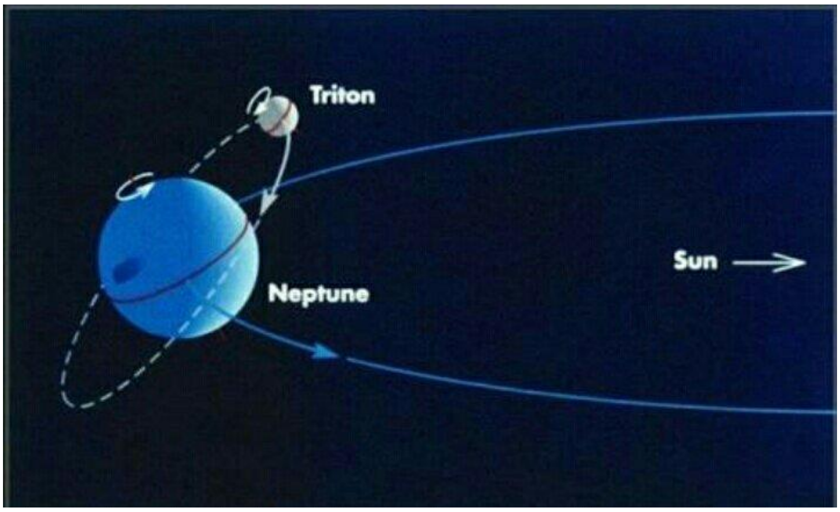



*Figure 8. Triton's orbit around Neptune. (anastrogeek,2017)*

In relation to the study *Non-sphericity of Triton's atmosphere* (Person, 2001), it stated that, even though Triton's atmosphere is non-spherical, there appears to be an apparent relaxed shape of the body, indicating that the atmospheric disturbances are not caused by its shape or any physical body distortions. Therefore, the consequences by a non-sphericity are not taken into consideration, thus Triton's body was assumed to be completely spherical when developing the equations and the mathematical approach during the previous work on this regard. (Core López, 2022)

After developing the long-term disturbing potential, the following equation was obtained:

$$R_2 = \frac{\xi^2 \mu_N}{a_N} \left(\frac{r}{a}\right)^2 \left(\frac{a_N}{r_N}\right)^3 x \left(e^2(\cos(i)+1)^2 \cos(2g+2h-2l_N) + e^2\, x\, (\cos(i)-1)^2 \cos(2g-2h+2l_N) - \frac{6}{5}\, x(\cos(i)-1)\left(e^2+\frac{2}{3}\right)(\cos(i) + 1)\cos(2h-2l_N) + (-2e^2(\cos(i))^2 + 2e^2)\cos(2g) + \frac{6}{5}(e^2+\frac{2}{3})(-\frac{1}{3}+(\cos(i))^2)\right)$$

*Equation 1. Long-term disturbing potential*

The terms obtained in Equation 1 refer to Neptune when indexed with an N, and to the CubeSat when in lower case. The most important terms to be taken into account shown in the above equation are the following: eccentricity (e), inclination (i), argument of the periapsis (g) and longitude of the ascending node (h), all of them belonging to data from the CubeSat's orbit.

Hence, the mathematical model to know the disturbing potential affecting the CubeSat, considering the gravitational attraction of Neptune, may be obtained using Equation 1.

The consequences that this potential has on the orbital elements of the CubeSat can be analysed using the Lagrange planetary equations, having as a result four non-linear differential equations:

$$\dot{e} = -\frac{a(1-e^2)}{G \cdot M \cdot m \cdot e}\frac{\partial R}{\partial T};$$

$$i = -\frac{1}{\sqrt{GMm^2 a(1-e^2)}\sin i}\frac{\partial R}{\partial \Omega} - \frac{1}{me}\sqrt{\frac{1-e^2}{GMa}}\frac{\partial R}{\partial \omega};$$

$$\dot{\omega} = \frac{1}{me}\sqrt{\frac{1-e^2}{GMa}}\frac{\partial R}{\partial e} - \frac{1}{\sqrt{GMm^2 a(1-e^2)}\tan i}\frac{\partial R}{\partial i};$$

$$\Omega = \frac{\dot{1}}{\sqrt{GMm^2(1-e^2)}\sin\imath}\frac{\partial R}{\partial \imath};$$

*Equation 2. Lagrange planetary equations.*

The subsequent steps for determining the variation of this orbit involve substituting the terms from Equation 2 into Equation 1, and then solving the integrals using mathematical software. Optimal orbits would be polar, as they provide superior surface coverage of Triton. Although a plot of the function was not included in this document, it was deemed unnecessary since it primarily involves substituting parameters.

It was been considered, however, to include the orbit discussion and development, at least briefly, so the simulation in **¡Error! No se encuentra el origen de la referencia.** can be correctly done and understood by the reader.

## 3.3 Power supply problem

The power system is broadly considered as the most critical throughout the different subsystems within a satellite. This significance stems from the potential consequences: should the power sources experience a malfunction, it could result in the mission's overall failure and the forfeiture of the spacecraft.

This problem is the main concern discussed during this project, evaluating the different possible sources such as batteries and solar arrays. It is further developed and explained in Chapter 4. However, the fundamental topic of analysis, development and research revolves around the radioisotope thermoelectric generators as main power source for the space probe. The comprehensive study of these components and all its research is fully talked about in Chapter 5.

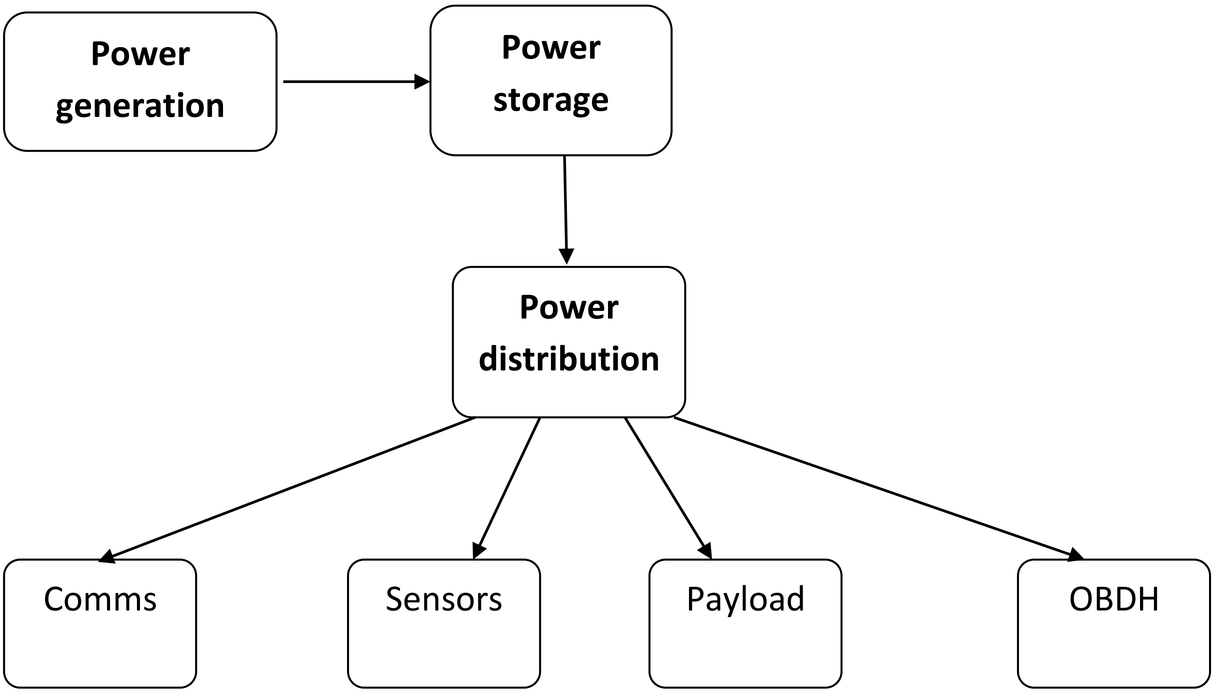
# Chapter 4. POWER SUPPLY SYSTEM

## 4.1 Power subsystem

During this chapter, the power subsystem as a whole is not only introduced, but also its importance is discussed, as well as the possible solutions manageable for such a constrained spacecraft of the kind of a cubesat.

The power subsystem within a cubesat, also called EPS (Electrical Power System), positions itself as an essential and crucial component of its operational architecture. Its primary role dwells in the production and distribution of electrical energy, essential for the subsistence of the satellite's diverse functions. Operating within the limitations enforced by the cubesat's small structure, the power subsystem requires a well-judged interchange of advanced energy conversion mechanisms, efficient power management techniques and precise thermal control strategies. This introductory exposition explores the complex architecture and significance of the power subsystem in the context of CubeSat missions, demonstrating its critical contributions to mission functionality and overall scientific objectives.

### 4.1.1 Power subsystem in a CubeSat



The block diagram shown above specifies the general complexity that the power subsystem withholds, as it manages and serves as source of every other system in the spacecraft.

For low Earth orbit and geostationary missions, the power subsystem does not impose an issue due to its closeness to the Sun and the radiation received from it, as it works as the main power source using solar arrays and backup batteries.

As an example of a mission away from Earth is the MarCO cubesat, which used solar arrays and backup batteries as its main power system, providing around 35 Watts of power at 1AU distance. (Directory, 2018)

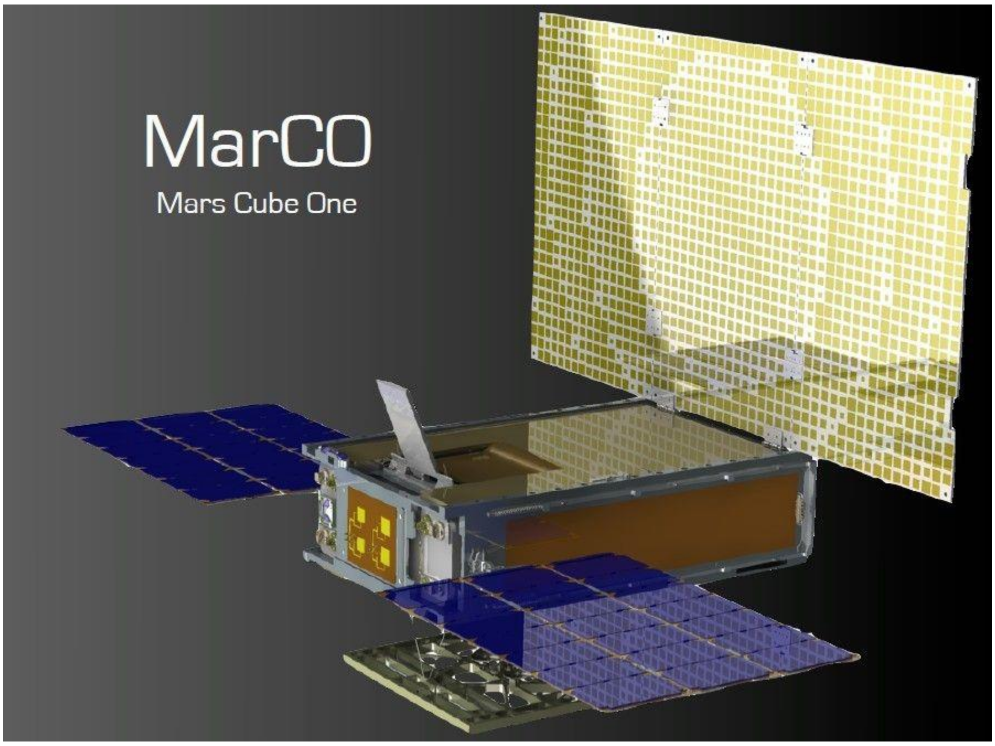



*Figure 9. MarCO CubeSat (Gunter's space page, 2021)*

However, the CubeSat discussed in this project is to partake and face a long-range, deep space interplanetary mission, much further into the Solar System. Thus, the power system implies a bigger problem that the space industry and aerospace engineers are to go up against for years to come, as the microsatellite industry keeps on expanding.

### 4.1.2 Possible solutions

As mentioned in 4.1, solar arrays are the most widely used form of energy source, especially for near-Earth missions. Despite the vast distance that is to separate the proposed CubeSat and the Sun, almost every known satellite and space probe is conformed with solar arrays in any way possible, regardless of the small amount of power that it may produce. Actually, in this case, a CubeSat utilizes and consumes much less power than a bigger satellite would, so for the CubeSat to have solar arrays is a given.

For calculating the size and therefore the power provided by these solar arrays, it is considered important to introduce the solar radiation intensity equation, and its progression with distance with respect to the Sun.

$$H_0 = H_{sun} \cdot \frac{R_{sun}^2}{D^2}$$

*Equation 3. Solar radiation intensity received*

Where:

- $H_0$ = Solar radiation intensity received ($W/m^2$)
- $H_{sun}$ = Solar radiation from the Sun ($W/m^2$)
- $R_{sun}$ = Radius of the Sun
- D = Distance of object from the Sun

For this mission, a hybrid between body-mounted and deployable solar arrays is chosen. This report does not investigate more extensively into the research, calculations or surface area of the solar arrays. Nevertheless, an estimation has been made to ease the comprehension of the system for the reader.

Being aware that this CubeSat's size is 16U (20 x 20 x 40 cm), and having concluded previously that body mounted and folding solar arrays had been chosen as the best choice, the solar arrays estimations are as follows:

| **Cell type** | **Typical size (cm)** | **Kg/m²** |
|---|---|---|
| GaAs | 4 x 6 | 2.8 |
| **Estimated solar array area (cm²)** | Estimated mass (kg) | Estimated power (W)* |
| 1920 | 0.5376 | 38.784 |

*Table 2. Solar arrays estimation. (Core López, 2022)*

It is important to remind the reader that the power produced by the solar arrays decreases with the square of the heliocentric distance, so the realistic estimation would be lower than 38 W. Thus, the estimation in Table 2 has been done for closer to Earth missions.

The secondary power system choice are batteries, to provide the necessary energy when the primary power system (solar arrays or RTGs in this case) is unable to do so. Hence, batteries are primarily used during eclipses or shadowed periods, and recharged during sunlight phases.

The best option for an optimal size (the main issue regarding these microsatellites) are lithium batteries, which specific provided approximate energy is between 90-150 W·kg/h. (Fortescue, 2011).

Conclusively, these two types of energy sources are a given for the CubeSat deep space mission developed in this report. From this point onward, a third and alternative energy source is researched and discussed as a possibility for the microsatellite industry, evaluating advantages and disadvantages, as well as carrying out a study about the current market and the experiments performed during the last years up until today.

## 4.2 Proposed solution

The limited effectiveness of solar arrays for deep space missions has been commented due to the low solar flux received. Consequently, the industry's primary matter for the next decade rests in the advancement of ground-breaking power sources suited for interplanetary missions involving small satellites.

In the context of an long, deep space mission, such as the one described in this document, solar arrays cannot function as the main energy source due to the inconveniences highlighted in section 4.1.2. Additionally, exclusive dependence on batteries is not recommended due to their limited life cycle, even when rechargeable.

Thus**,** radioisotope thermoelectric generators (RTGs) are introduced. These components base their operation on the thermoelectric effect, making it possible to generate electricity between materials when a temperature difference is maintained. (Fortescue, 2011)

This system is further and deeply discussed as the main objective of development in Chapter 5. Every advantage and drawback is mentioned, as well as its whole functionality, different experiments and how it would fit inside a microsatellite.

# Chapter 5. RADIOISOTOPE THERMOELECTRIC GENERATORS

In this chapter, the radioisotope thermoelectric generators are introduced as the proposed solution for the power subsystem problem in a long range deep space cubesat mission.

Several points are to be discussed and researched, such as the process of operation that these systems have, as well as advantages, possible upgrades, how it would fit in a cubesat system and a market study to include it in a mass and financial budget.

## 5.1 Functionality and operation

During this subchapter, the physical principle followed by RTGs is described, followed by the description of the components that form these power systems.

### 5.1.1 Physical functionality principle

Space exploration missions require the presence of consistent and steady power systems that can supply both electricity and warmth to spacecraft. A validated and reliable means of generating power is through the utilization of Radioisotope Power Systems (RPS). Among these, the Radioisotope Thermoelectric Generator (RTG) stands out: an advanced space-based nuclear power technology that transforms heat into electric energy without the requirement of any mobile components. (NASA, 2020)

In 1821, the German scientist Thomas Seebeck discovered a very interesting property of physics: that both metals and a few compounds are good conductors of electricity and heat. He learnt that, when combining two of these materials and applying heat and cold to each of their ends, an electrical voltage is produced across them. Meaning, electrons would stream from the heat source to the colder side, and, the greater the temperature difference between the hot and cold sides, the greater the power generated. This way of producing electricity is now called the Seebeck effect.

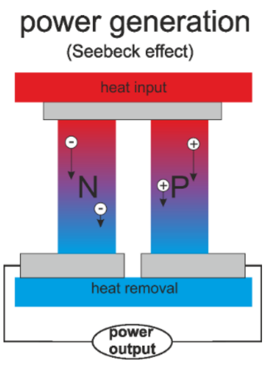



*Figure 10. Seebeck effect. (ResearchGate, 2015)*

At this point, it is important to introduce the term *thermocouples*, which are the main components used to take advantage of the previously defined Seebeck effect in RTGs. These electrical devices or components consist of two different electrical conductors, forming an electrical junction.

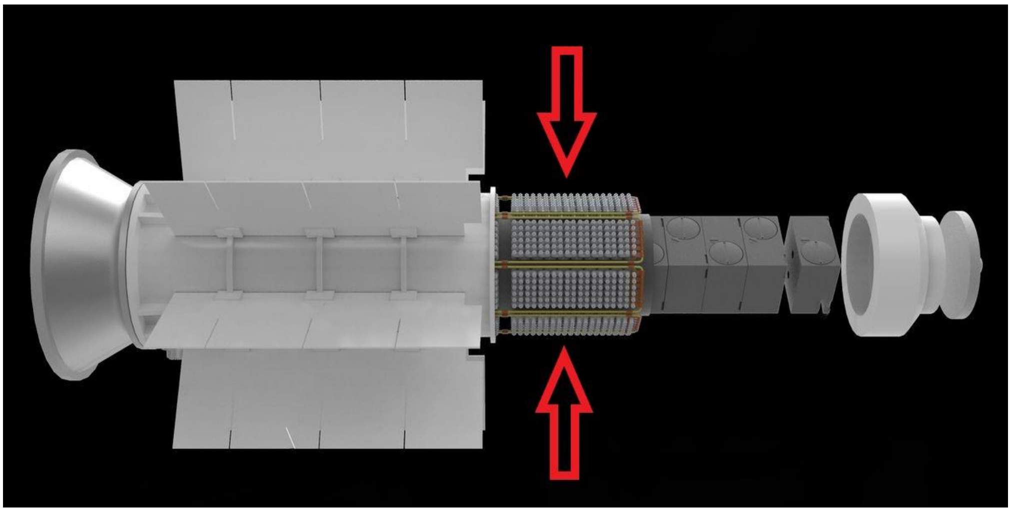


*Figure 11. RTG interior. (NASA RPS, 2023)*

As it can be seen in Figure 11, thermocouples are added in RTGs as hundreds of small pairs of junctions to perform the Seebeck effect previously described.

### 5.1.2 RTG description and usage in space

During the last years of aerospace industry development, it was learnt that a space nuclear power system could be developed and used in space missions as the main energy and electricity source, which are called Radioisotope Power Systems (RPS), or Radioisotope Thermoelectric Generators (RTG) in this case.

These type of power systems, which follow the previously described physical principle to generate power, produce electricity by the heat emitted from decaying radioactive isotopes through the thermocouple devices. RTGs mainly consist on two subsystems: a thermal source and an energy conversion system. The thermal source is the radioactive fuel, and the conversion system used in all RPS units used up until now, is a thermoelectric converter.

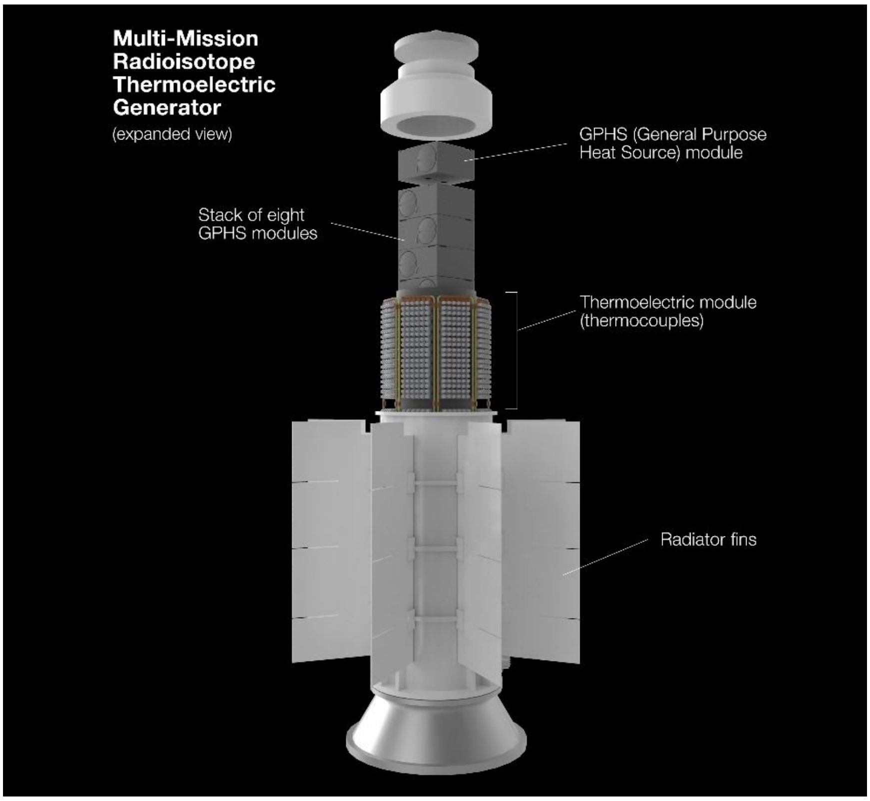



*Figure 12. MMRTG model. (NASA RPS,2023)*

As mentioned, Radioisotope Thermoelectric Generators utilize the heat from the natural radioactive decay of, specifically, plutonium-238, as plutonium dioxide. The huge temperature difference between the heat emitted from this fuel and the cold space, generates an electrical current following the Seebeck effect described.

As it can be seen in Figure 12, RTGs are formed by different components which perform important tasks for this effect to be optimized and appropriately applied. The diverse components are (NASA, 2018):

- **GPHS modules**: This stack of modules is the main heat source were the nuclear material is placed, called General Purpose Heat Source. The materials used for this modules are, among others, ceramic, metal cladding, graphite and carbon-fibre, mainly for protection purposes in case of ground impact or accidental re-entry. The RTG fuel (the isotopes) is stacked in this modules, thus the importance of its protection and safety.
- **Thermocouples**: These are the solid-state metallic junctions in which the temperature difference is applied for the electrical current to be generated.
- **Radiator fins**: These components are used for heat dissipation (heat sink). The design of these radiators are such as it provides a larger thermal gradient between the core and the outer layers, even though it means having a mass penalty when compared to other possible radiator designs.

The widespread configuration of an RTG is fairly simple and clear. It consists of two essential components: radioactive decay-prone fuel and a wide array of thermocouples with the objective of transforming heat into electrical energy. The radioactive fuel is placed behind the layer of thermal insulation, while modules containing thermocouples are distributed along the RTG's sides.

As mentioned, the fuel inside the RTG that decays radioactively and provides the heat has to be chosen taking into account that it must accomplish the following characteristics (Stanford, 2013):

- Ability to produce high energy radiation. The isotope selected should be able to release sufficient energy during the decay process for it to be an effective source of the thermoelectric conversion. This characteristic is the least strict out of the four noted, as many isotopes include it.
- Tendency to produce radiation decay heat. For the heat generation to be effective in such a condensed system as an RTG, it has to arise in the constraint space of the device walls. Thus, the radiation absorption length is to be short, alpha being the radioactive decay which has the shortest one. Meaning, the highest quantity of heat would be produced by alpha radiation decay.
- Possession of long half-life for continuous energy production. It is important to take into account that the re-fuelling chances of the RTG are null as it is to be isolated in deep space, therefore the fuel and isotope chosen must comply with the possibility of producing energy for long periods of time.
- Large heat power-to-mass (or density) ratio. In this case, the size efficiency is taken into account, not only for the creation of a compact RTG, but especially for the mission designed in this report, as a CubeSat is crucially structured and sized constraint, so the amount of fuel (therefore, the chosen isotope) to produce the required energy does not compromise the mass, weight or any other element of the spacecraft.

After analysing the previous factors, scientists came to the conclusion that the best isotope choice would be Plutonium-238. Other isotopes were also a possibility: Strontium-90 and Americium-241. Conclusively, Pu-238, as the best option for the RTG fuel, has already been used in almost two dozen space missions, from the Cassini mission to both Curiosity and Perseverance Mars rovers. However, a shortage of plutonium is being noticed in the nuclear power systems for space industry, so the other two isotopes must be taken strongly into account as best options for future space missions, including the one designed for this project.

| Isotope | Half-life (years) | Power / Mass ratio (W/Kg) |
|---|---|---|
| Plutonium-238 | 87.7 | 0.54 |
| Strontium-90 | 28.1 | 0.46 |
| Americium-241 | >432 | 0.114 |

*Table 3. Isotope data comparison.*

In Table 3, it can be seen why, taking into account those important values, plutonium-238 was selected by scientists as the best possibility. However, the potential substitute for this isotope is to be Americium-241. Despite it's lower heat power-to-mass ratio (density), its long half-life would able interstellar probes to perform their missions for decades.

Following the selection of the fuel and its consolidation, it is subsequently placed within a multi-layer containment structure typically referred to as the heat source. Several of this containment layers are used, to ensure minimal risk of radioisotopes dispersion if an accident during launch or re-entry occurs. The European Space Agency (ESA) has a strict safety policy and restrictions regarding nuclear power sources, which is further discussed in 5.5.

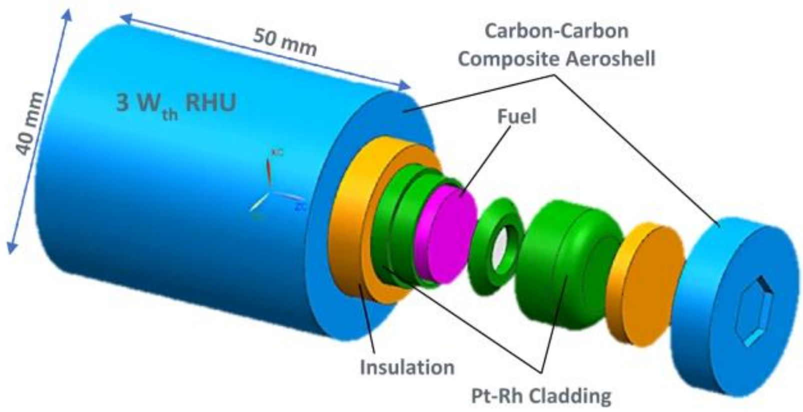



*Figure 13. Schematic architecture example of heat source. (Ambrosi, 2019)*

Once the isotope composed fuel is discussed, other important element is to be discussed: thermocouples. These elements, as mentioned earlier during this chapter, are of utmost importance, as they receive the heat collected from the heat distribution blocks (which are connected to the fuel that radioactively decays), to then convert that heat into electricity. Thermocouples should be composed of materials with low thermal conductivity, in order for there to be a strong temperature gradient, and thus produce higher electrical power. The most common materials used for the thermocouples in RTGs are: bismuth telluride (BiTe), lead telluride (PbTe) and silicon germanium (SiGe). These elements capture the heat generated by the RTG's isotope fuel, inducing a significant temperature difference owing to their limited thermal conductivities. Consequently, they generate electric currents that the RTG channels to the components of the spacecraft that require power.

*Figure 14. RTG installed in the New Horizons probe.(NASA, 2019)*

The components described are the main and basic components that a Radioisotope Thermoelectric Generator must be composed of. However, there are different prototypes and models that have been developed and used for space missions and explorations, which are further discussed and analysed in 5.3.

## 5.2 Other missions

In this subchapter, a few space missions that used radioisotope thermoelectric generators as their main power source are mentioned and described, getting deeper into each model in 5.3.

For the last fifty years, both NASA and ESA have developed individual and collaborative missions for space exploration, which go from Mars exploration to the outskirts of the Solar System and beyond. One of the main issues encountered has always been the power system, as scientists and engineers investigated deeply in order to optimize this system: make it more efficient and optimize energy supplied to the spacecraft.

Despite there being a few possibilities and options to tackle this problem, such as solar panels and batteries (view Chapter 4), the research and advancement on new innovative approaches and solutions has not ceased.

| Probe | Mission purpose | RTG model | Mass (kg) | Electrical Output (nº/We) |
|---|---|---|---|---|
| Cassini | Saturn exploration | GPHS | 56 - 58 | 3/275 each |
| Galileo | Jupiter exploration | GPHS | 56 - 58 | 2/275 each |
| Perseverance | Mars exploration | MMRTG | <45 | 1/110 |
| Curiosity | Mars exploration | MMRTG | <45 | 1/110 |
| New Horizons | Pluto / Kuiper belt | GPHS | 56 - 58 | 1/245 |
| Pioneer-10 | Outer planets | SNAP-19 | 13.6 | 4/40 each |
| Pioneer-11 | Outer planets | SNAP-19 | 13.6 | 4/40 each |
| Voyager 1 | Deep space | MHW | 37.7 | 3/150 each |
| Voyager 2 | Deep space | MHW | 37.7 | 3/150 each |

*Table 4. Previous missions using RTGs. (Jhaveri, 2021)*

In Table 4, it can be seen the different space missions during the last years and decades that have used radioisotope thermoelectric generators as the main power source and the models which were used for each one. The subsequent subchapters go deeper into some of these missions, and the RTGs used for each one, describing them and their main characteristics and performances. The missions themselves and the other elements of the spacecraft are omitted, as are not considered the main objective of this chapter.

### 5.2.1 Cassini - Huygens

The Cassini-Huygens spacecraft was developed between different space agencies, including NASA, ESA and ASI. The purpose of the mission was to explore and orbit Saturn and deploy the Huygens probe on Titan, and was composed by a dozen different science instruments.

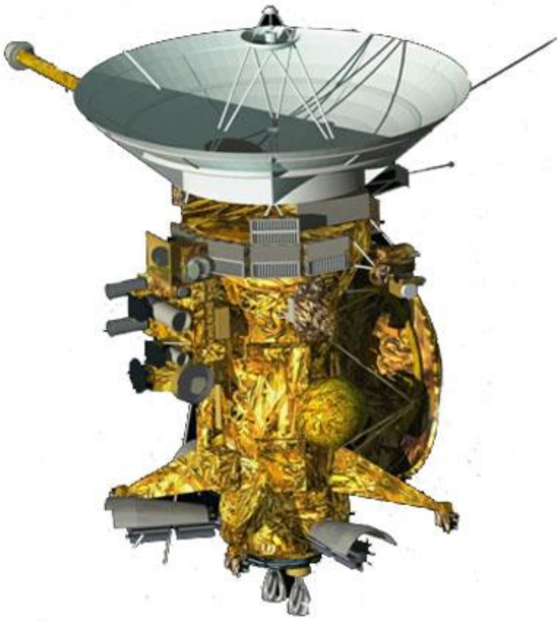


*Figure 15. Cassini-Huygens probe. (NASA, 2019)*

For this mission, three GPHS-RTG models were used, each one of them using 10.9 kg of Pu-238. Each of these models produced almost 300 Watts of power, continuously feeding every equipment and instrument on the spacecraft. The power supplied came down to 210 Watts after over a decade. On the bottom part of Figure 15, the radioisotope thermoelectric generators can be observed.

### 5.2.2 New Horizons

This spacecraft, launched in 2006, was the first one to have the purpose of exploring Pluto and the Kuiper Belt, and was developed by NASA as part of the New Frontiers program.

It used a single unit model of GPHS-RTG, providing over 240 Watts of power at the moment of its launch. By the time it reached Pluto in 2015, the power produced was reduced to 202 Watts. It contained almost 10 kg of Plutonium-238 fuel.

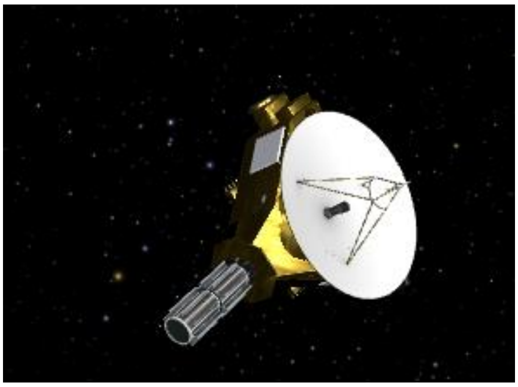


*Figure 16. New Horizons RTG.(NASA, 2021)*

The single GPHS-RTG used can be seen at the bottom of Figure 16. In this case, only one unit of a radioisotope thermoelectric generator was used due to the fact that this mission not only required less power as it had much less scientific instruments, but also because of its much smaller size in comparison with the Cassini-Huygens probe.

### 5.2.3 Pioneer–10 and Pioneer–11

The first unmanned missions that used radioisotope thermoelectric generators as main power source were the Pioneer 10 and 11 probes, launched in 1972 and 1973 respectively. Both spacecrafts were virtually identical, with a similar purpose but a few slight changes. The main mission of Pioneer 10 was to fly by Jupiter, and reached said planet at the end of 1972. It was composed by several instruments to study Jupiter's environment. It lost communications with Earth in 2003, at an approximate distance of 80 AU from our planet.

On the other hand, Pioneer 11 was launched on April 1973, with the intent of performing a fly-by of Saturn, which was achieved in 1979, becoming the first spacecraft to observe the ringed planet.

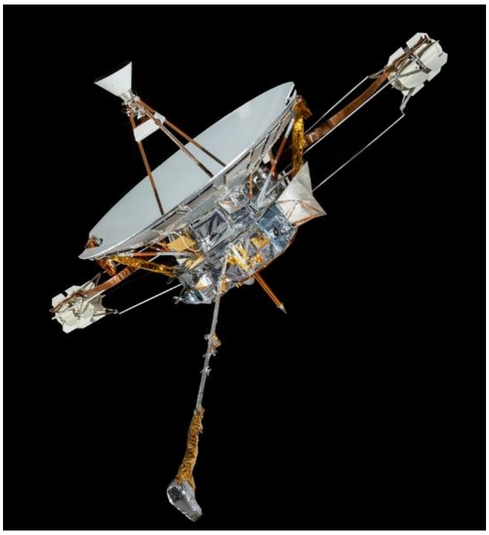


*Figure 17. Pioneer 10 and 11 spacecraft model. (National Air and Space Museum)*

Both these probes used four units of SNAP-19 RTG models, with heat from twelve RHU (radioisotope heater units), positioned on the on the left and right tips of the spacecraft, as can be seen in Figure 17. These power systems provided 155 Watts of power at the time of launch, decaying to 140 Watts in transit to Jupiter.

### 5.2.4 Voyager 1 and Voyager 2

The Voyager missions were two interstellar probes launched in 1977, with the objective of reaching the outskirts of the Solar System. Both probs reached and passed the ice giants, Uranus and Neptune (the first ones of doing so), in 1985 and 1989. These probes are the only artificial satellites that have gone into deeper interstellar space, far away from our Solar System, and both are still sending important information to Earth after over 40 years. Figure 18 is used to show the reader the huge and vast distance that these probes have travelled.

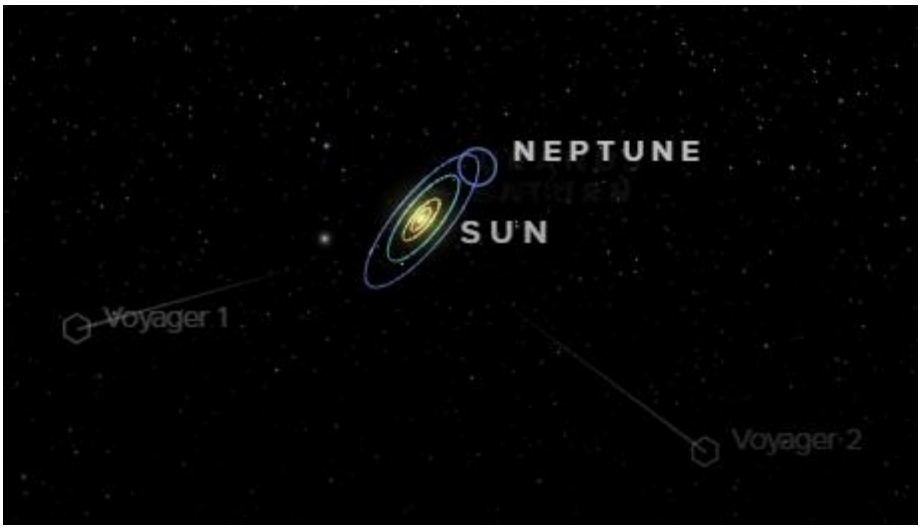



*Figure 18. Distance of Voyagers probes. (NASA, 2023)*

Both spacecrafts use three units of the MHW-RTG models mounted on a boom. They generate around 150 Watts of power each unit, and weighting less that 40 kilograms. The RTG model used for this probe can be seen on the bottom right of Figure 19.

*Figure 19. Voyager spacecraft. (NASA, 2019)*

As it can be noted, different models of radioisotope thermoelectric generators were used for more or less similar missions (deep space), so the first approach when choosing the best option and design for the CubeSat mission developed for this project, would be to take into account the different performances that these models can provide, apart from their efficiency and possible optimization for power production and mass constraint. This is further discussed in 5.3.

### 5.2.5 Mars Perseverance and Curiosity

Curiosity mission landed on Mars in 2012, and it is a robotic vehicle which mission was to explore the martian land, climate and geology. As of today, this rover is still active on Mars's surface, so it has been working for over a decade. As its predecessors missions and the first landers on Mars, Viking 1 and 2, this rover is powered by an RTG, the latest RTG model developed in this case: MMRTG, which stands for Multi-Mission Radioisotope Thermoelectric Generator. *The Vikings missions were launched in the 1970s and used SNAP-19 RTGs, same model as the Pioneers.

*Figure 20. Mars Curiosity rover. (NASA, 2019)*

Curiosity's MMRTG was fueled by 4.8 kilograms of plutonium-238, and provides 110 Watts of power, coming down to 100 Watts after 14 years.

Perseverance mission is a Mars mission rover which was designed to explore an specific crater in martial land, called Jezero. It landed successfully on Mars on 2021, and to this day is still working and active.

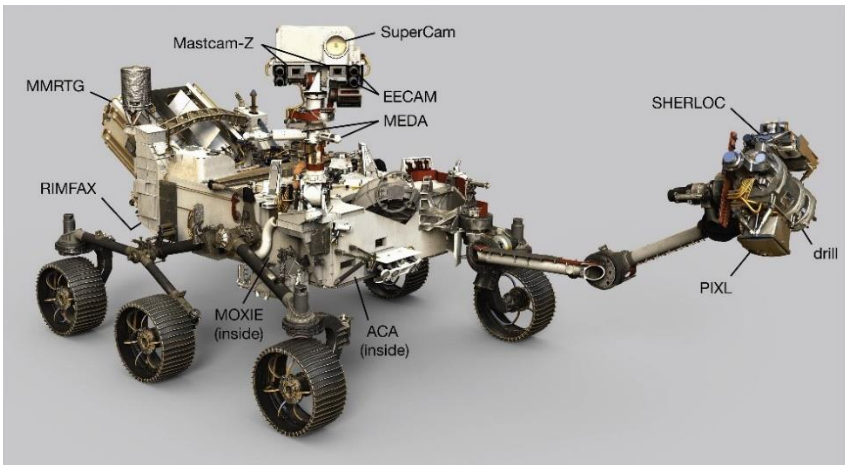



*Figure 21. Mars2020 Perseverance rover components.(NASA, 2020)*

Perseverance's and Curiosity0s designs are very similar, only differing in the scientific equipment installed and a few other characteristics. Perseverance rover also used the state-of-the-art MMRTG for power supply, using 4.8 kilograms of Pu-238 and providing around 110 Watts of power.

These two rover missions are mission because, even though they are not deep space missions, the newest and most recently developed MMRTG are used, so it is important to take that into account when approaching the variety of radioisotope thermoelectric generators available nowadays.

The Galileo mission could also be mentioned for this subchapter among the other deep space missions. However, it was discarded as it utilized the same RTG model as the Cassini, providing more or less the same performance. It is considered important to note that the various space missions are mentioned mainly to bring up the diverse RTG models used in the past and in the present, and how crucial they have become for deep space and long-time missions (e.g, Voyager missions).

## 5.3 RTG models and data

Once the explanation on how a radioisotope thermoelectric generators work and the previous missions where these systems were used in has been developed, for this subchapter the different existing models are to be analysed, pointing out the different parameters that are perceived as most important for this kind of system. Only RTGs used and developed for space purposes are compared, and the focus has been projected to main four RTGs models, as they have been considered the most important ones. Also, the different thermocouples possibilities are mentioned, described and compared, in order to understand the choice made in 5.5.1.

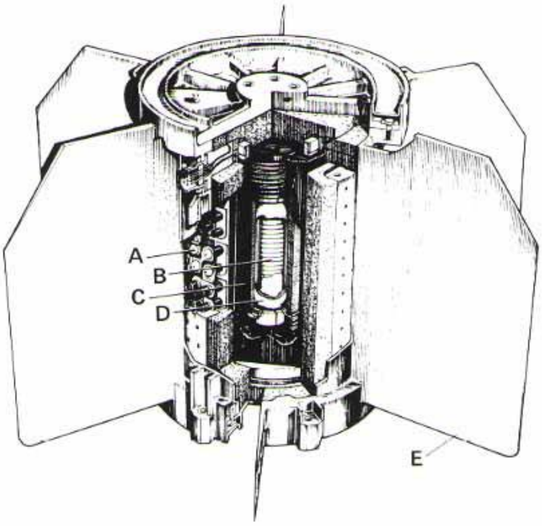
There exist different types and models of radioisotope power systems that have been utilized for space purposes and missions, as it was discussed in 5.2. The main purpose of all of them are the same: to be the main power supply for the spacecraft elements and equipment. Some space missions only used RTGs as the power source, but most of them also had solar panels like an inherent characteristic of a satellite.

Following is the description of each RTG model used in space, with the design and thermocouples used, as well as the performance and other parameters that are considered important to take into account.

### 5.3.1 SNAP-19

This RTG system belongs to a NASA developed program named Systems Nuclear Auxiliary Power, which purpose was to experiment with radioisotope thermoelectric generators and space nuclear reactors.

Several models of the SNAP type of RTG were developed during the 1960s, ranging from SNAP-1 to SNAP-27. The main differences amongst them were the isotopes used and the electrical power output provided by each of them, improving over time, or even optimizing them.




A THERMOELECTRICS
B FUEL CAPSULE
C REENTRY HEAT SHIELD
D FUEL DISCS
E HEAT RADIATING FINS

*Figure 22. SNAP-19 RTG model cutaway.(NASA Image Collection)*

As noted in 5.2, this model of RTG was used for both the Pioneer missions, but it was also used in other space missions such as the Mars landers Vikings 1 & 2, however with a small modification making this model be called SNAP-19B.

The SNAP-19 prototype used Plutonium-238 as the fuel, mainly using n-type 2N-PbTe and p-type TAGS-85 thermoelectric elements, and is the lightest developed to date. This model, among the ones that are to be researched and described in this chapter, is the one that provides the lesser electrical output. For the Voyagers, four units of these systems were needed to amplify the power provided to the spacecraft. (DOE, 1973)

As it can be seen in Figure 22, the cutaway of the design of the RTG shows the main elements that these systems currently still have, such as the heat radiating disc and the fuel capsule. It is important to keep in mind that the design is about forty to fifty years old, and still the same approximate design is being utilised for space missions, speaking highly of the performance that this power sources may provide in a spacecraft.

### 5.3.2 MHW-RTG

The multi-hundred watt RTG model was used for the Voyager missions, and was specifically developed to face the challenges encountered when planning and developing deep space missions, such as the Voyager, to the outer planets of the Solar System, far away from the Sun.

Deep space missions encounter several power problems such as the direct proportion between power provided and communications bandwidth, so more power is necessitated, as well as a longer lifespan.

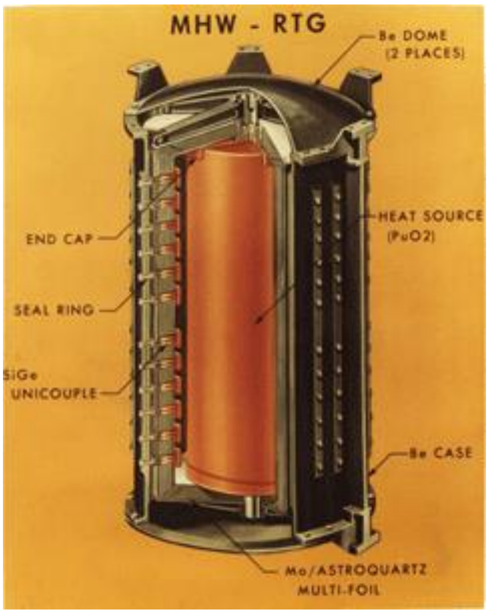



*Figure 23. MHW-RTG model cutaway. (Beyond nerva, 2020)*

This RTG model also used Plutonium-238 as the fuel, and utilized silicon-germanium thermocouples for the thermoelectric generators, that were up to 312 units which were attached to the outer casing of the RTG.

### 5.3.3 GPHS-RTG

The GPHS-RTG, General Purpose Heat Source was created by the General Electric Space Division. It is considered the most successful RTG design, as it powered major scientific missions: Ulysses, Galileo, Cassini and New Horizons.

Its design came after the MHW-RTG, as even more power was needed for the previously mentioned missions. The revolutionary design of this RTG consisted on building a modular radioisotope thermoelectric generator around a radioisotope heat source, the GPHS. A total of 18 modules could be stacked to provide the heat source.

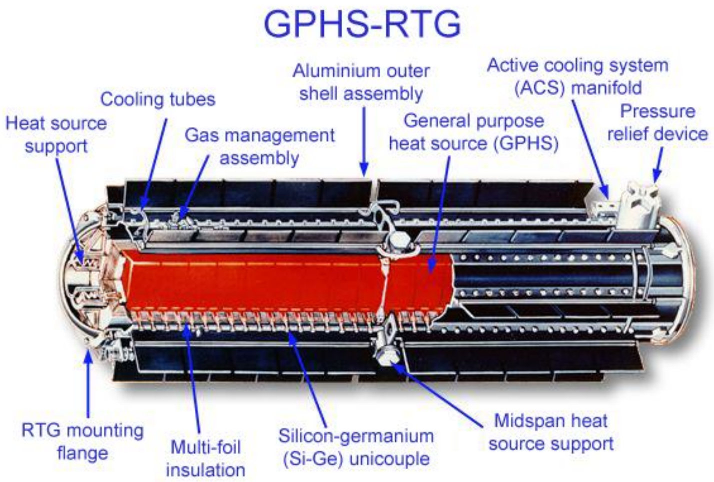



*Figure 24- GPHS-RTG model cutaway. (DOE/NASA/JPL, 2006)*

This model also uses Pu-238 as the fuel that gives off the heat to produce the required energy, and the casing and protection material surrounding the modules is mainly graphite, and also uses silicon-germanium thermocouples, called unicouples.

This model evolved to the currently used MMRTG, but the GPHS module is still used, being the essential building block of the RTG developed by NASA.

### 5.3.4 MMRTG

The multi-mission radioisotope thermoelectric generator is the latest model of the radioisotope power systems developed to this date. It takes characteristics from its predecessors, such as the GPHS module still being in use for this design.

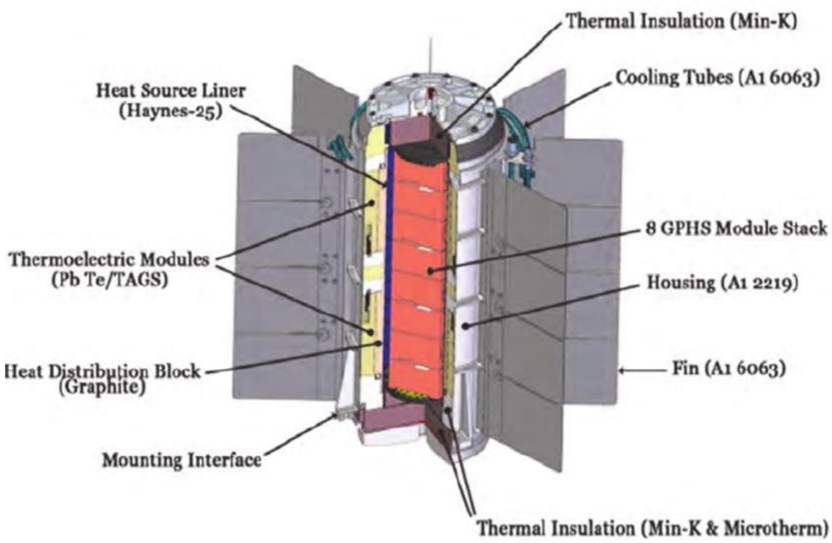



*Figure 25. MMRTG model cutaway. (ScienceDirect, 2020)*

It has been only been used for the Curiosity and Perseverance Mars rovers, and is still yet to be utilized for further, deep space missions. It also uses Plutonium-238 as fuel and incorporates PbTe/TAGS thermoelectric couples, leaving behind the silicon-germanium semiconductor combination which is no longer in use for this purpose and systems.

As it can be seen, the models described utilize the same fuel and have a similar approach and design, the main differences regarding the size, the number of radiator fins, the thermocouples used and some other features. The comparison between the considered main characteristics and parameters of these systems can be seen in Table 5. (DOE, 1973) (NASA, 2023)

| Model | Isotope | Diameter (m) | Mass (kg) | Thermocouple | Electrical output (W) | Power/mass (W/kg) | BOL* System efficiency |
|---|---|---|---|---|---|---|---|
| **SNAP-19** | Pu-238 | 0.12 | 13.6 | PbTe/TAGS | 40.3 | 2.9 | 6.2 |
| **MHW-RTG** | Pu-238 | 0.397 | 37.7 | Si-Ge | 160 | 4.2 | 6.5 |
| **GPHS-RTG** | Pu-238 | 0.422 | 56 | Si-Ge | 300 | 5.2 | 6.3 |
| **MMRTG** | Pu-238 | 0.64 | 45 | PbTe/TAGS | 110 | 2.8 | 6.3 |

*Table 5. RTG models comparison.*

*BOL → Beginning of life.

Analysing the data obtained to complete Table 5, it is to be noticed the huge difference made and the optimization of the electrical output between the first model designed, SNAP-19, and the models that followed. Also, despite the SNAP-19 model being the smallest one, its efficiency does not stand that back from the later developed prototypes.

It can also be observed that the GPHS produces the highest electrical power, almost doubling its predecessor. However, it is the heaviest due to the amount of fuel and the number of GPHS modules stacked.

Lastly, the newest and current model of RTG in use, MMRTG, has a much lower power to mass ratio and produces much less electrical output. Nonetheless, its development and design was made to be fit in much smaller spacecrafts, rovers in this case, and for closer to Earth missions, so not a huge amount of power was needed, compared to the Cassini mission, for example.

## 5.4 Advantages and drawbacks

During this chapter of the project, the different advantages and disadvantages that using radioisotope thermoelectric generators have for space missions, especially noticing that they indeed are nuclear energy systems, which poses a somewhat ethical and moral question.

When researching and developing a system to be applied to a space mission, the mission and system engineering process entails the evaluation of said system, thus putting in perspective if the usage of that system regarding not only the spacecraft in which it is to be implemented, but also the outcome of doing so, and every possible consequence or result that it may have, positive and negative.

As it has been discussed during 33Chapter 5, radioisotope thermoelectric generators are built to endure several years, even decades of useful working live. When this system was first invented during the 50's, and then developed and perfectioned during the following decades, its design was mainly focused on being compact and durable, to make it an ideal power source for remote operations.

Its composition and configuration was made to be able to withstand and resist the unrelenting environment and freezing temperatures of space, especially deep space, where the Sun does not influence as an actual hot source as it does for the closer planets. The advancement on the materials used to encapsulate the RTGs radioactive interior used as heat source for the Seebeck effect to take place, has only made an improvement on safety, durability, and mass and weight issues that are faced during space missions, also making this energy source extremely lightweight.

Radioisotope thermoelectric generators do not need maintenance, as they have no moving parts, and, having been used on several long-term and long-range space missions, have proven to be remarkably reliable, as they are still working on missions such as the Voyager, which were launched almost forty years ago. This last matter proves that RTGs provide further longevity, as its lifetime depends on that of the fuel source, with the power only shrinking due to the radioactive decay and anything else affecting it. (Dubois, 2020)

When it comes to the drawbacks that radioisotope thermoelectric generators hold, the most mentioned is the one pointed out in the first paragraph of this subchapter: environmental risks. The usage of radioactive material as the source, Plutonium-238 for every case up until now, has been the main reason for scientists and society in general to question the employment of these energy systems. However, even though this radioactive material obviously emits radiation, its main form is in alpha particles, which can easily be obstructed using a sheet of paper. Even human skin is able to block it.


*Figure 26. Replica of NASA's plutonium pellet. (Member, 2020)*

So, the higher risk of this radiation to become a real danger, is when it takes the form of pulverized particles than can be inhaled, causing serious damage to organs. However, the radioactive material used in RTGs is combined with oxygen to create plutonium dioxide, in the form of grains or pellets, which are harder to inhale. It can be conclusively said that the actual environmental disadvantages of the usage of a radioactive material such as plutonium are rather insignificant or of little importance.

Currently, the main inconvenience that the usage of radioisotope thermoelectric generators is facing, is the shortage of Plutonium-238. In 2014, NASA had about 35 kilograms of this material accessible, and, as in previous space missions about 11 kilograms were used on each one, nowadays NASA only has this radioactive material available for two or three more missions.

Conclusively, the main concern regarding the future of RTGs is the fuel used due to the lack of Plutonium-238.

### 5.4.1 Safety

When developing such a power source as RTGs, safety concerns arise due to the usage of the radioactive fuel employed: exposure risk, unintentional release and dispersion of radioactive materials, and unauthorized access to the radioactive material.

Therefore, when facing space missions, there always exists the possibility of a mission accident, followed by a potential release and dispersion of the radioactive material into the environment, making the utilization of radioisotope power systems somewhat controversial.

However, for the last decades, these systems and their safety have been perfectioned and optimized to reduce and minimze the risks arised in case of an accident. For example, as mentioned in 5.2.1, the Cassini-Huygens probe's RTG was kept in high-strength blocks made of graphite, also surrounded by a layer of iridium metal.

The latest radiosiotope thermoelectric generators used in the Mars Science Laboratory mission, the MMRTG model, included safety features such as the iridium metal cladding, graphite sleeves protecting fuel clads, and carbon-fiber forming the aeroshell model. (Jhaveri, 2021)

Regarding accidents occurred with a radiosiotope power system inside, two can be mentioned:

- Transit 5BN-3: This accident took place on 1964, during the launch of the navigational satellite, and it was caused by a computer malfunction, resulting on the satellite feailing to reach its designated orbit. The RPS used for this mission was a SNAP-9A RTG, which reentered Earth's atmosphere and burnt completely, and was intended to due to its design. The accident's outcome was the release of about 20000 Ci. (Jhaveri, 2021)
- Nimbus B1: This meteorological satellite had its launched terminated on 1968, as commanded by the range safety office, due to the erratic ascending launch vehicle. Both the vehicle and the satellite were shattered, however, the RTG used, which was the SNAP-19B2 model, remained intact after impact, and the plutonium heat sources were recovered and reused in another mission. (Jhaveri, 2021)

### 5.4.2 Improvements

The main advancement that should take place to improve radiosiotope thermoelectric generators and continue its development, is the research and experiments using another isotope as fuel.

Several studies have been done regarding this matter, and their main focus for a new isotope to be used, is Americium-241. As described in (Several authors, 2013), this isotope would be used as fuel when combined with oxygen, resulting in americium trioxide in the form of ceramic pellets. In that same study, a few experiments were performed, concluding that using an americium fuelled RTG does not much the performance when plutonium is used for the same purpose ay higher power output levels. However, its power levels may be acceptable for a mission such as the one described in this project, as the power needed is to be much smaller than for other deep space missions discussed during the document.

Another space radioisotope power system that was developed and studied as an energy converter, was the advanced stirling radioisotope generator, ASRG, a project designed by NASA, which was canceled in 2013 due to budget constraints. However, it is considered important to mention that using an Stirling engine as a dynamic thermoelectric convertor in RTGs is a real possibility for future space missions, as this way of converting energy is has proven to be more efficient for higher power output levels (100 We and above).

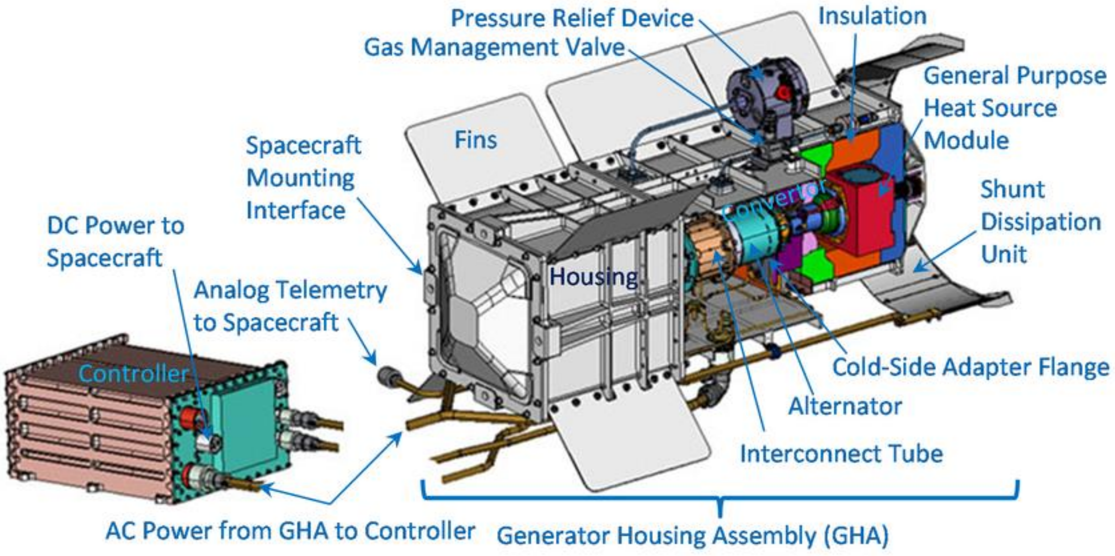



*Figure 27. ASRG flight design. (Schmitz, 2015)*

In Figure 27, the cutaway of the design of an Advanced Stirling Radioisotope Generator can be seen with all its components. It consists of one GPHS module adjacent to each Stirling engine convertor on its heat side. The heat is converted to electricity using and advanced Stirling convertor (ASC), which is a free-pyston Stirling engine with an integrated alternator that allows this conversion, from the reciprociation motion of the pyston to actual electrical output.

| | |
|---|---|
| **Power at launch** | 145 – 155 We |
| **Mass** | 21-23 kg |
| **Specific power** | 6.4 – 7.4 We/kg |
| **System efficiency** | 29 – 31 % |
| **Environments** | Deep space and martian surface |
| **Lifetime** | 14 years operating |

*Table 6. ASRG estimated performance. (DOE, 2008)*

Therefore, continuing the study of both a new isotope as the Americium-241 as well as retaking the possibility of using a dynamic thermoelectric convertor such as the Stirling engine, may be a crucial path to follow in order to improve even more the capacities and performances of RTGs.

## 5.5 RTGs adaptability

During this subchapter, the possibility of applying this power system to such a constraint spacecraft as a CubeSat is researched and discussed, developing a series of parameters and performances that the RTG may have for certain conditions of mass, diameter, etc.

It is important to note that this is merely hypothetical, retrieving some data from other studies made that can be similar and be an approximation of how an RTG would work in a CubeSat, and chosen to be applicable as a possible outcome of the usage of this power system.

### 5.5.1 Hypothetical RTG design

Firstly, it is important to recall the dimensions of the desired CubeSat for this mission, as well as the mass budget expected and approximated. It is also needed an estimate of the power that the CubeSat may need, and verify and check if the RTG proposed may achieve the power output required.

Several constraints need to be taken into account when approaching such a challenging matter:

- The fuel to be used.
- Size limitation.
- Mass limitation.
- Radiation shielding.
- Safety encapsulation.

| Parameter | CubeSat value |
| --- | --- |
| **Mission objective** | Triton observation |
| **Payload** | Imager |
| **Size** | 20 x 20 x 40 cm, 16U |
| **Mass** | 9 – 11 kilograms |
| **Expected Power Output** | 40 – 50 Watts |
| **Lifespan predicted** | 5 years |

*Table 7. CubeSat main values.*

The required power output is based on an approximation taking into account the power needed of the electrical components that the CubeSat is to have for each subsystem, and taking into account the most common output power values of other cubesats. The lifespan of the CubeSat mission is given in order to give the reader an idea of the duration of the usage of useful power output (when transmitting data to mothership, mainly).

| Subsystem | Specific components (#) | Estimated power required (W) |
|---|---|---|
| **Payload** | Gecko imager | 2.6 – 4.5 |
| **Thermal Control** | Temperature sensors (2) | 0.5 - 1 |
| **TT&C** | Patch antenna<br>Transceiver | 5<br>5.5 |
| **On-board processing** | Micro-processor | 3.6 |
| **ADC** | Sun sensors (3)<br>Star trackers (3)<br>Reaction wheels (3) | 3 |
| **Propulsion** | Canted thrusters | 12 - 15 |
| **Total estimated power output required** | --- | 32.2 – 37.2 |

*Table 8. CubeSat components estimated power output.*

The components used to estimate the values are obtained from (GOMSpace, 2023) and (Satsearch, 2023). For the thermal control, MLI and radiators have been disregarded also as no power is needed.

Now, pointing to the fuel used, **Plutonium-238** as the isotope to be combined with oxygen is the most obvious choice due to it having proven itself reliable and efficient in deep space missions previously. The **pellets** produce should be very small, but not minuscule for safety reasons. The shielding for the pellets would be made of **graphite**, but tungsten is also considered a great possibility (University of Bristol, 2015)

As silicon – germanium thermocouples were dismissed several years ago, the thermocouples chosen would be **PbTe/TAGS**, following the MMRTG model. The radiator fins would be made of **aluminium**, but carbon-fiber could be better for mass restrictions.

Taking into account that the Pu-238 provides around 0.57 W per gram, only about 65 grams of Pu-238 would be needed as fuel (for peak power output), therefore dismissing both solar panels and the rechargeable Li-Ion batteries as the main and backup power sources. The considered size of the RTG, given that the CubeSat dimensions are 40 x 20 x 20 cm, should be between 1U-2U, this is 10x10x10 cm (both for cubicle or cylindric form). Furthermore, even a smaller sized RTG could be made and input one on its side of the CubeSat instead of placing just one, as RTGs are placed on the outside of the space probe. It also has to be considered the usage of canted thrusters, taking some space in the outskirts of the CubeSat. Nevertheless, with this information at hand, the insulation and casing masses are easier to approach.

It is a hard task to approximate a mass budget for the proposed RTG, as it should be made by hand an experimented with, especially when referring to the usage of casing, shielding, etc. However, an estimation and assessment of a possible range of mass budget can be made from the information of COTS products and other studies made for smaller RTGs. The mass approach is as shown in  Table 9:

| Component | Mass (kg) |
|---|---|
| **Fuel mass** | 0.065 |
| **Thermocouples** | 0.160 |
| **Insulation** | 0.0576 |
| **Casing** | 2.26 |
| **Radiators** | 0.864 |
| **Total mass expected** | 2.5 – 3.5 |

*Table 9. Proposed RTG mass approximation.*

The values in the previous table are all **approximations** (not for the fuel mass). The number of thermocouples and the exact measures of the encapsulation and the radiators is yet to be known, but the measure of 10x10x10 cm has been used, as well as 5 mm wide insulation and 6 radiators of 20 cm long and 10 cm wide each.

Lastly, it is worth mentioning a new study being held and developed by NASA NIAC, about radioisotope thermoradiative cell power generator, where a thermoradiative cell is used as thermal convertor. This opens the doors to even lighter power systems for deep space in the future.

Conclusively, adding an RTG as the main power subsystem would become an excellent choice regarding power output. Though it would raise the mass budget, it would only increase in about 1 to 1.5 kilograms, as both the solar panels and batteries would be dismissed from the CubeSat and would lessen the mass in 1.5 kilograms. The actual efficiency and real power output of such a small radioisotope thermoelectric generator is yet to be experimented with, being a great next step for future works regarding this whole mission and project.

# Chapter 6. ORBIT SIMULATION

This chapter has been dedicated to show a simple visual simulation of the orbit that is to be followed by the CubeSat developed for this mission, utilizing the General Mission Analysis Tool (GMAT) software.

Firstly, the description of the software is described, followed by the several possibilities in which it may be used, and the whole development of the orbit simulation for this specific mission, and how the orbit transfer would be performed from Neptune to Triton.

## 6.1 General Analysis Mission Tool software

The General Analysis Mission Tool software, generally known as GMAT, is a software system used for developing, optimizing, simulating and estimating trajectories and missions, and was developed mainly by NASA, with the collaboration of the Air Force Research Lab and private industry. (Conway, y otros, 2010)

*Figure 28. GMAT software logo. (NASA, 2016)*

This software is open-source, and supports all kinds of missions, from low Earth orbit to deep space. It is widely used for real mission support, engineering projects and general public engagement. (NASA, 2016)

This system has important first steps when modelling the missions, which include creating resources such as spacecraft, propagators, estimators and optimizers.

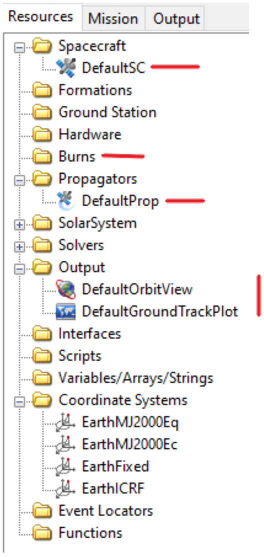



*Figure 29. GMAT resource tree example. (Elaboración propia, 2023)*

As it can be seen in Figure 29, the resources tree for the mission developed can be seen (in this case, there is a default mission inputted when GMAT is first opened). There is the option of editing the default spacecraft and changing its different parameters, such as attitude, mass, the visual model and so on.

It is possible to add either impulsive or finite burns, which are used when performing an orbit transfer, such a Hohmann transfer, when needing to achieve certain orbit insertion.

Propagators also are an important element, which combines an integrator and a force model. In the end, a propagator contains the physical model of the space environment used to model the motion of the spacecraft.

As shown in Figure 30, the configuration of the propagator implies selecting the appropriate numerical integrator, on the left, and the force model, on the right, which means choosing a gravity model, atmosphere model, bodies that affect the spacecraft, etc.

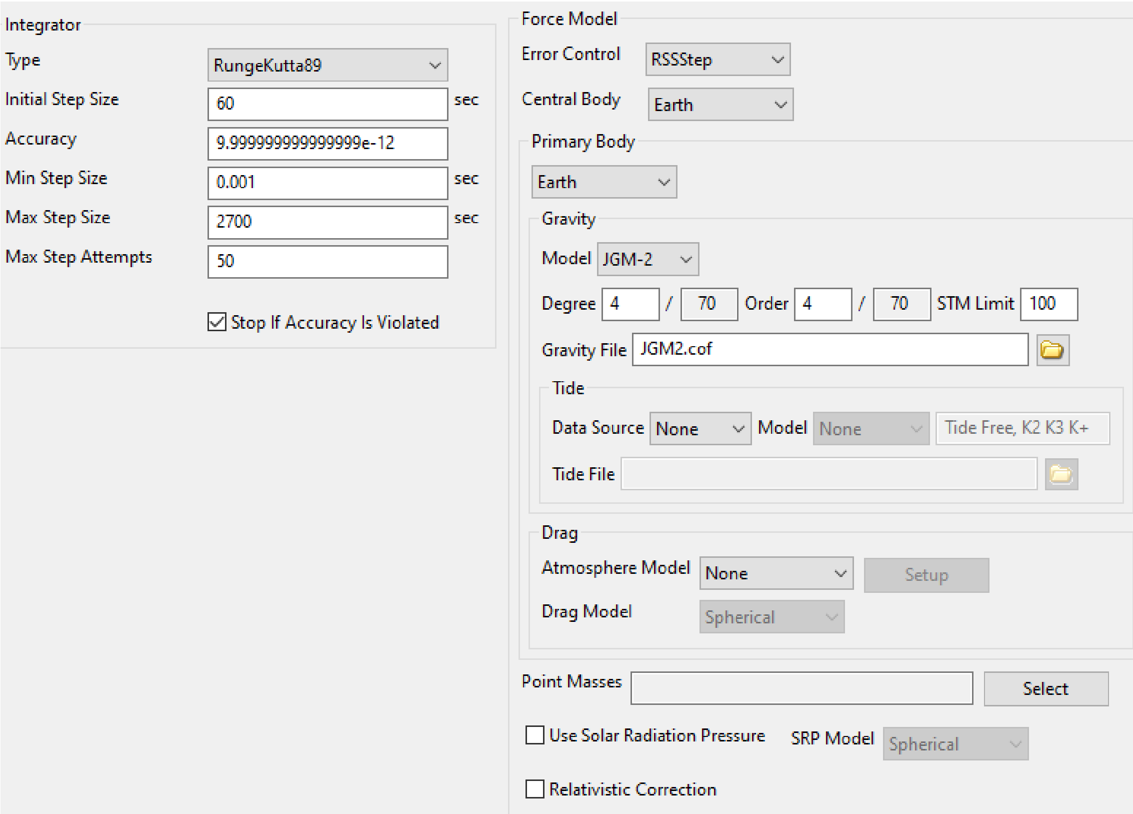



*Figure 30. Propagator parameters. (Elaboración propia, 2023)*

It can also be added hardware such as specific thrusters, tanks or power systems; different solvers such as the Yukon optimization solver or a differential corrector, and so on and so forth. There are multiple options when adding elements and components to the spacecraft and the whole mission, so the General Mission Analysis Tool is very complete.

Depending on the resources that have been added and edited, the mission tree would have different elements and processes on it, as it represents the mission sequence. In the example integrated by default by the software, only a propagator is added, so that mission only consists on a propagation of the spacecraft. A more complete mission sequence is shown in subchapter 6.2, to make it easier for the reader to understand how that mission tree sequence works.

Lastly, when the mission development is finished, having added all the necessary resources for the spacecraft and all the steps that the mission sequence is to perform, the mission can be run, using the button pointed and shown in Figure 31.

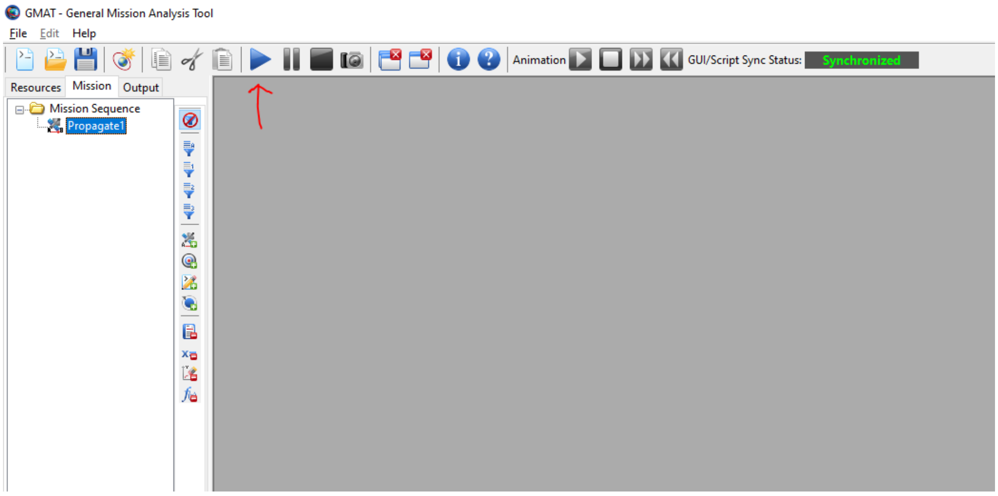



Figure 31. Mission run button. (Elaboración propia, 2023)

Once that blue run button is clicked, two different views are shown: Orbit view and Ground track plot.

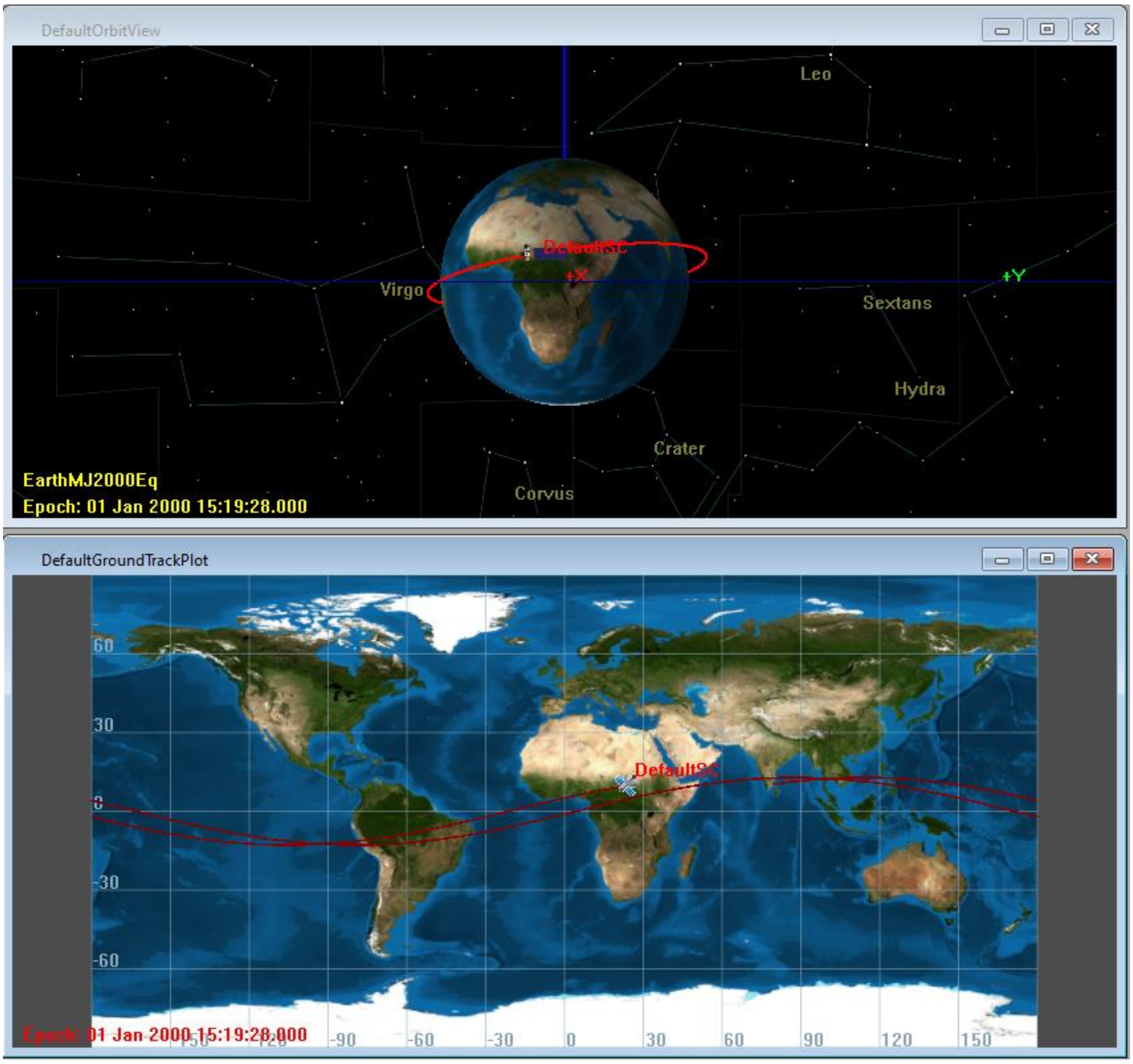



Figure 32. Mission views. (Elaboración propia, 2023)

The orbit view can be freely moved around with the mouse, showing the trajectory of the mission, a default orbit around the Earth in the case of this example. On the left, down corner of the orbit view, both the coordinate system chosen and the epoch can be seen. The epoch is what indicates the time and date for a corresponding orbit state.

On both views, the full animation of the mission can be seen, making it faster or slower, to fully visualize the complete trajectory and mission, and the behavior of the spacecraft.

This brief introduction to the software used is considered important to show all the possibilities that the General Mission Analysis Tool gives us, providing different choices for any type of mission.

## 6.2 Simulation development

During this chapter, the steps made to perform the orbit transfer from Neptune to Triton, which is the one that the CubeSat will follow for this mission being launched from the mothership orbiting Neptune, are shown and explained.

### 6.2.1 Step 1: setting inertial frames

The first step was to input the desired inertial frames and coordinate systems, Earth and Neptune in this case. This is done on the resources tree.

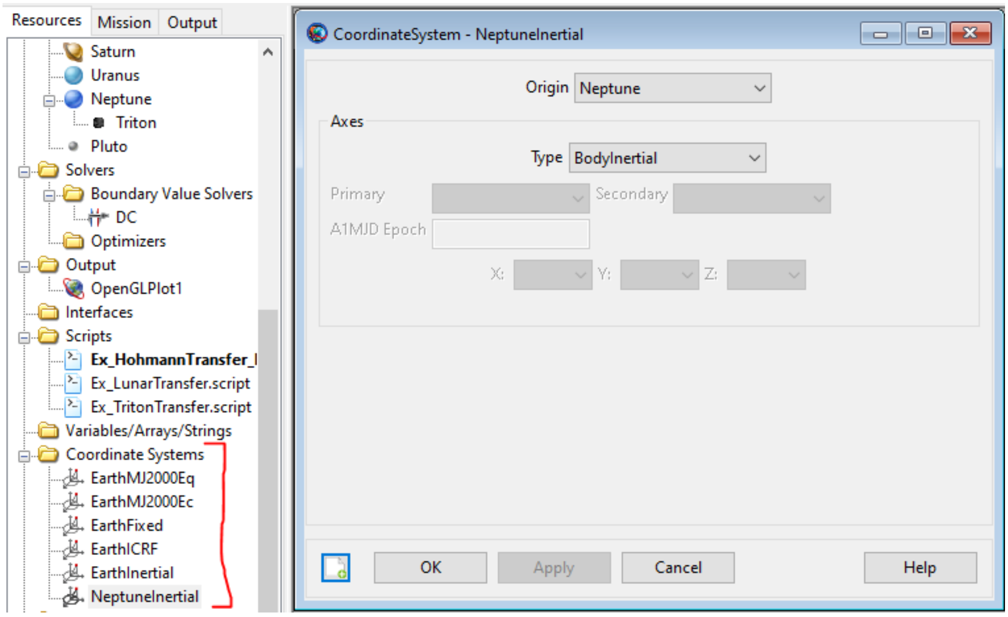



*Figure 33. Setting inertial frames and coordinate systems. (Elaboración propia, 2023)*

### 6.2.2 Step 2: Configuring CubeSat parameters and data

Next, the spacecraft parameters had to be included. UTC Gregorian epoch format was used, as well as a Keplerian state system. On the right side, the elements of the desired orbit to reach are inputted, following that of Triton's (some data is unknown).

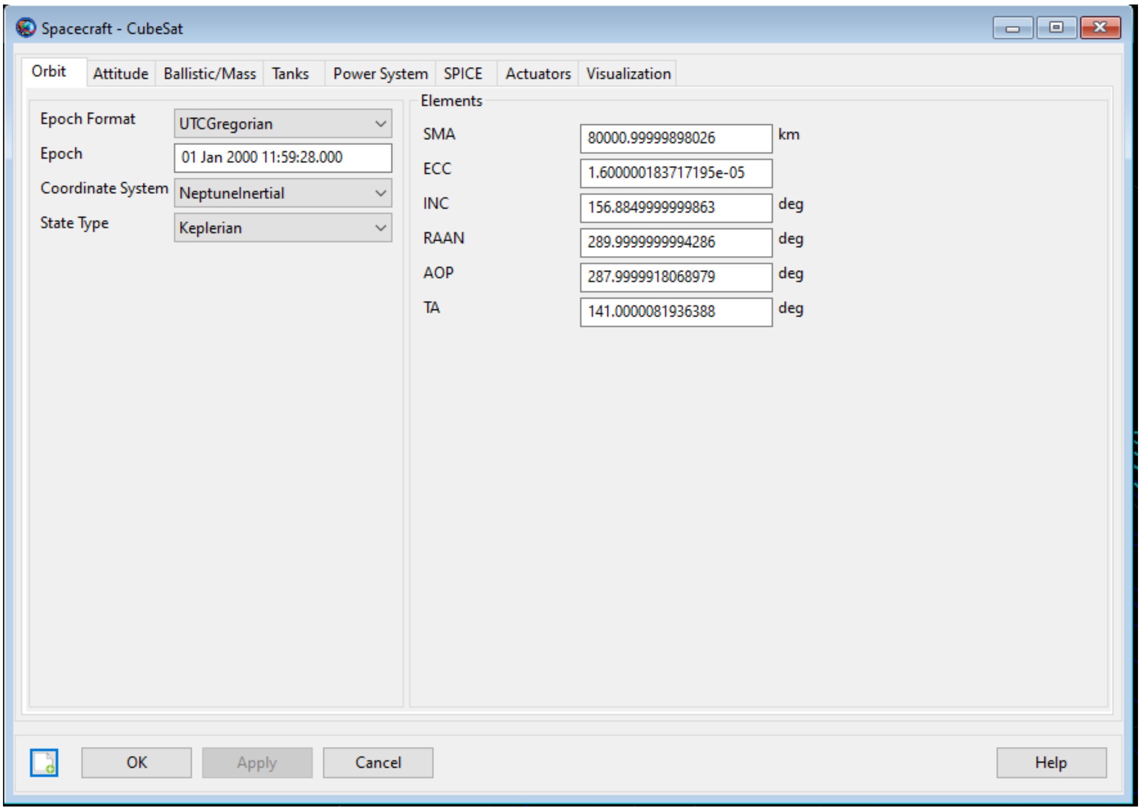



*Figure 34. CubeSat's orbit desired parameters. (Elaboración propia, 2023)*

### 6.2.3 Step 3: Adding impulsive burns: TOI and GOI

The burns are made to perform certain manoeuvres to achieve a certain target, a Hohmann transfer in this case. The target sequence is used in the mission tree in order to solve for specific and precise manoeuvre values. Basically, the software must be told the manoeuvres and the conditions to be satisfied for each (orbit radius, eccentricity,…).

This first burn is a TOI, transfer orbit insertion, and is to be added on the resources tree, as well as the GOI, geosynchronous orbit insertion.

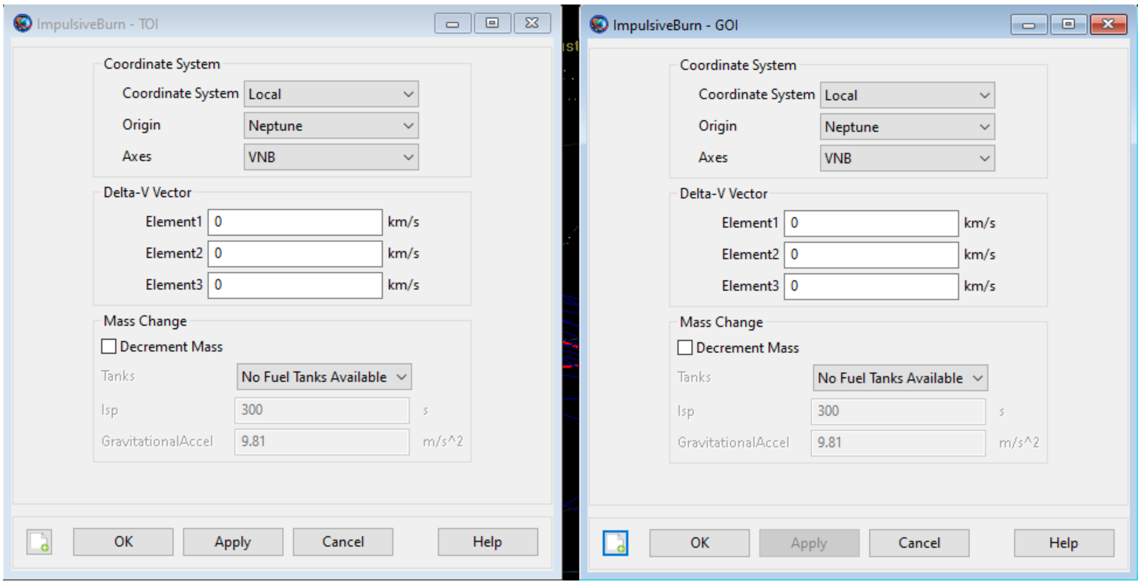



*Figure 35. Impulsive burns added (TOI and GOI). (Elaboración propia, 2023)*

### 6.2.4 Step 4: Configuring the propagator

In this step the propagator is configured. It is important to note that Neptune was used as a point mass.

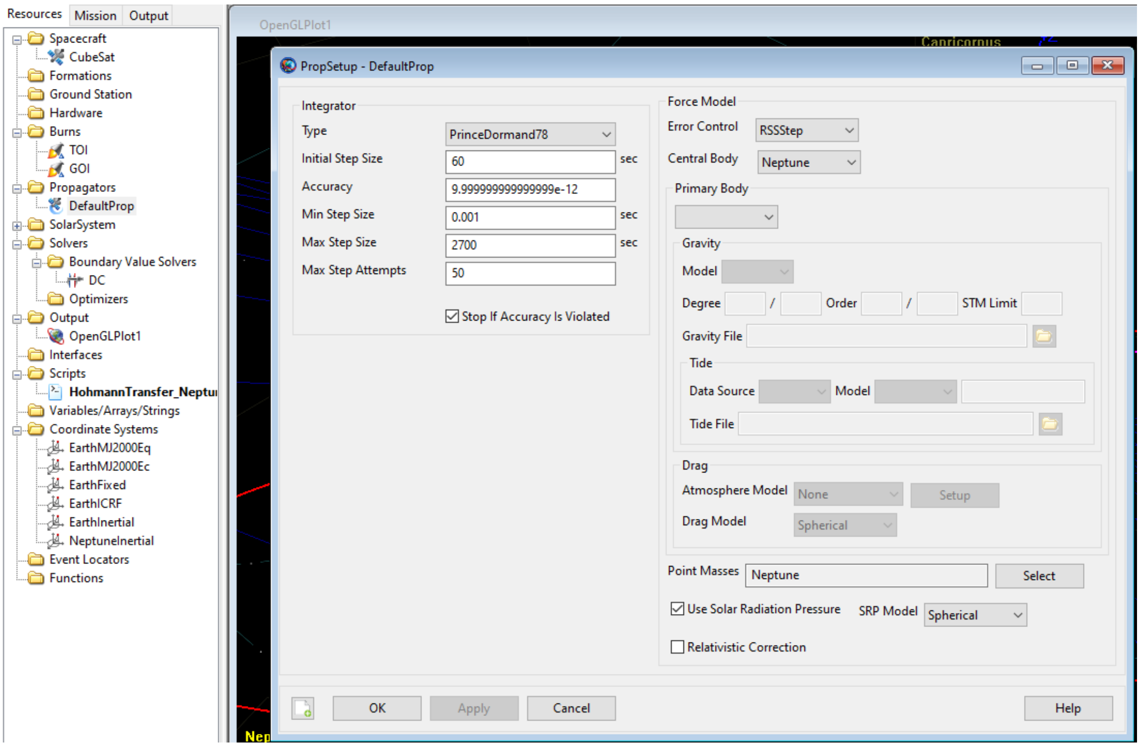



*Figure 36. Propagator parameters. (Elaboración propia, 2023)*

### 6.2.5 Step 5: Solver configuration

When defining a target sequence, a solver must be used, so a differential corrector is to be inputted in the resource tree for the target sequence to work properly.

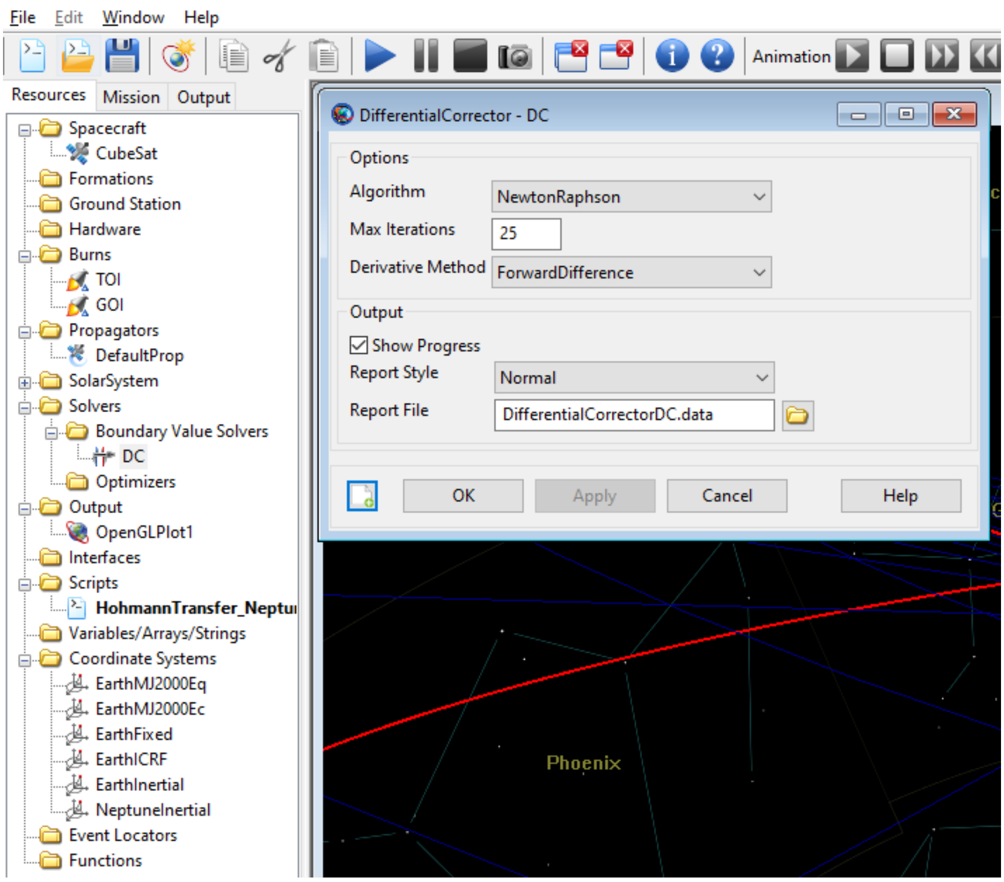


*Figure 37. Differential corrector solver. (Elaboración propia, 2023)*

The differential corrector is used as a numerical solver for two-point value problem, normally given during a target sequence, like in this case.

### 6.2.6 Step 6: Configuring mission tree

Moving to the mission tree, the first element to configure is the propagator defined previously in the resources tree. The propagation of the spacecraft to the perigee is the first step on the mission sequence.

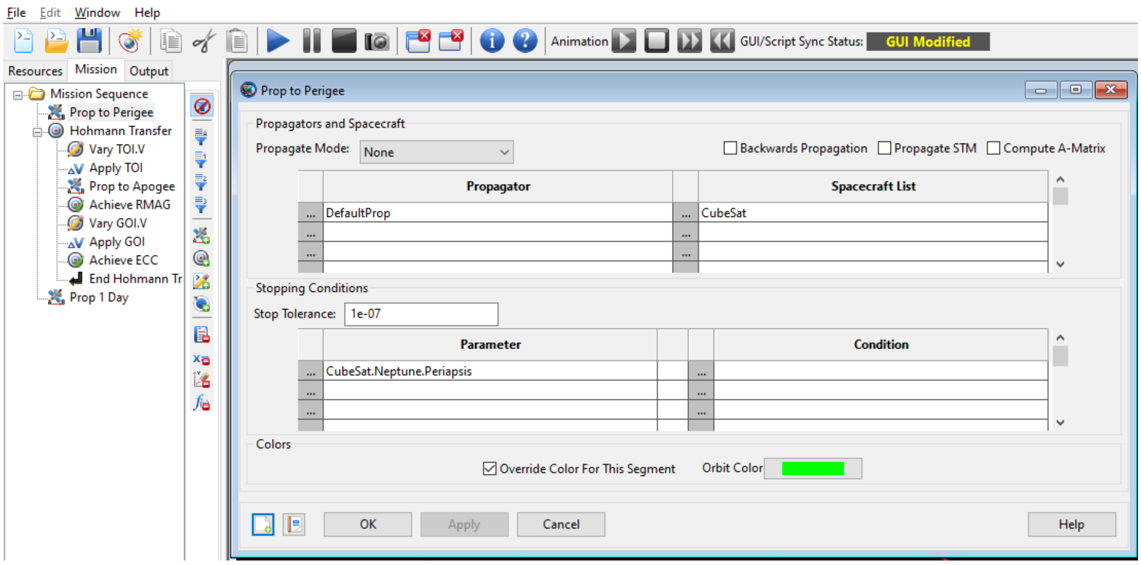



Figure 38. Propagate to perigee. (Elaboración propia, 2023)

### 6.2.7 Step 7: TOI

Next, the TOI vary and apply commands are inputted in the mission tree.

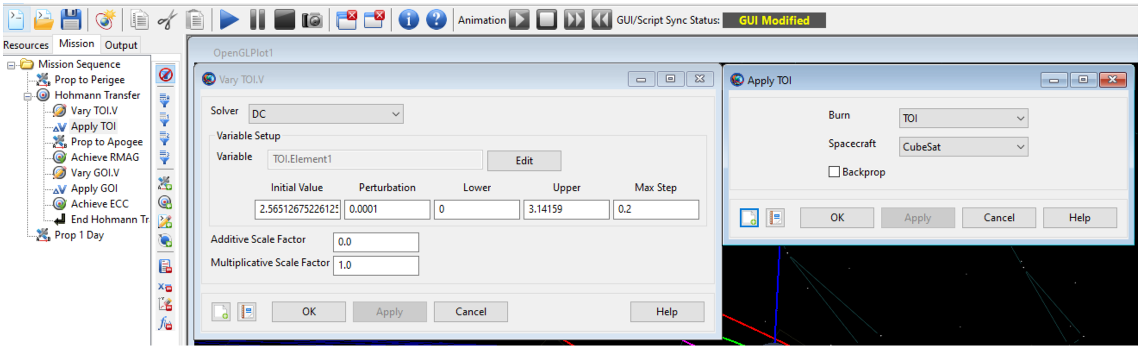



Figure 39. TOI parameters. (Elaboración propia, 2023)

This represents both the impulse given, and to the object that it is given (CubeSat in this case).

The next step in the sequence of the mission tree, is propagate to the apogee of Neptune, which is the target for which the TOI was used in the first place.

In Figure 40, the propagator to use is selected, which is the one previously defined in the resources tree. Afterwards, the parameters to stop said propagation is inputted, which is reaching Neptune's apogee in this case.

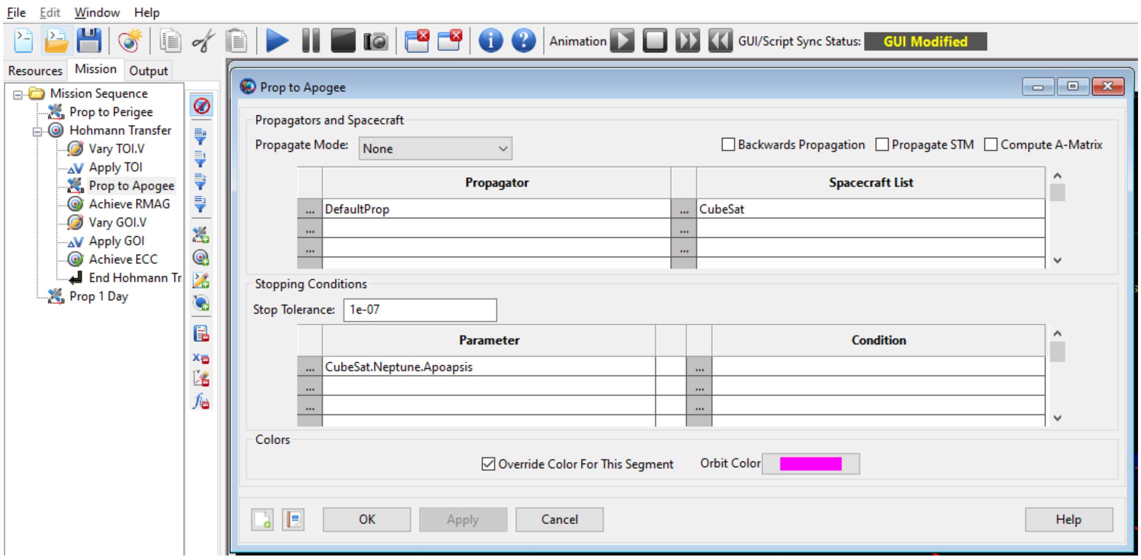



*Figure 40. Prop to Neptune's apogee. (Elaboración propia, 2023)*

Following the propagation to apogee, another target is introduced: achieve RMAG, which is the magnitude of the orbital position vector expressed in the coordinate system chosen in the CoordinateSystem field. This property, and many other object properties (RAAN, apogee, perigee, DEC, ECC, …) are given in GMAT, so it is possible to input a certain object property value (or more than one) as objective for the target sequence and proceed with the solver, like in this case. In Figure 41, the RMAG achieved at the end of the TOI sequence is shown.

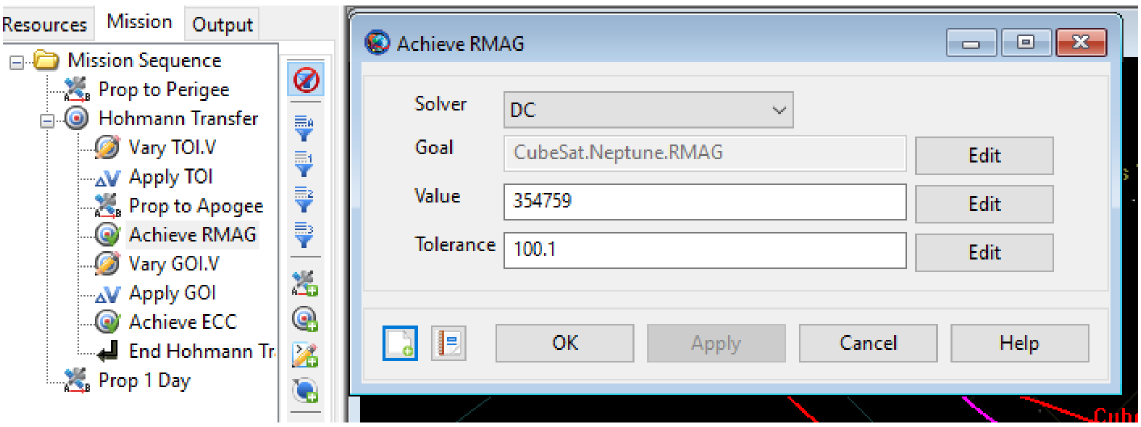



*Figure 41. Position at end of TOI. (Elaboración propia, 2023)*

### 6.2.8 Step 8: GOI

In this step, the sequence follows the process of applying the GOI impulsive burn, which is the second one after the TOI.

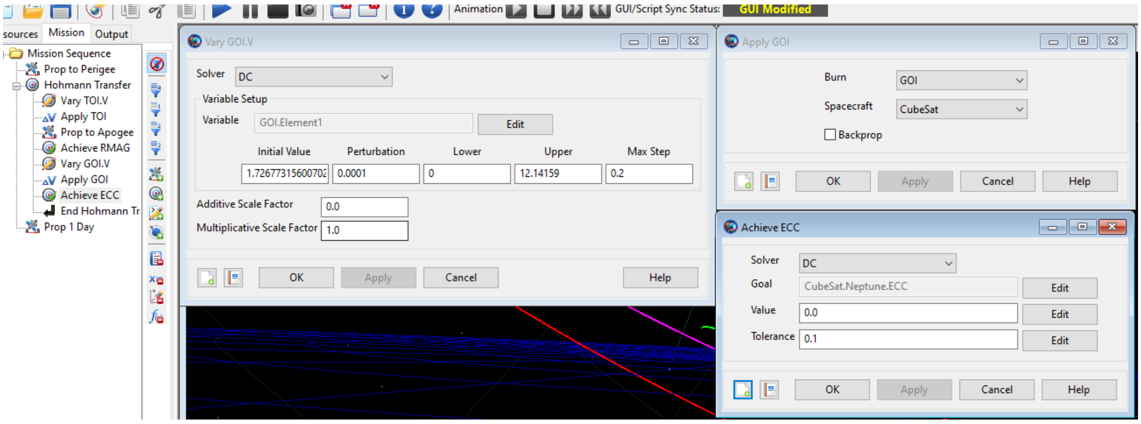



*Figure 42. GOI parameters. (Elaboración propia, 2023)*

Here, the same procedure as for the TOI is followed: the values of the impulsive burns and to the object to which they are applied, CubeSat in this case.
On the bottom right of Figure 42, the desired target of GOI is shown: achieve certain eccentricity, which for Triton is basically zero (1.600000183717195e-05).

When that target sequence is followed and the objective values are achieved, the Hohmann transfer ends, and GMAT proceeds to the next step in the mission sequence tree, which is propagating the CubeSat for one day in the final achieved orbit.

### 6.2.9 Step 9: Run the mission

When the run mission button on the top of the toolbar is clicked, the solver proceeds to iterate and luckily converge on a solution to achieve the desired orbit.

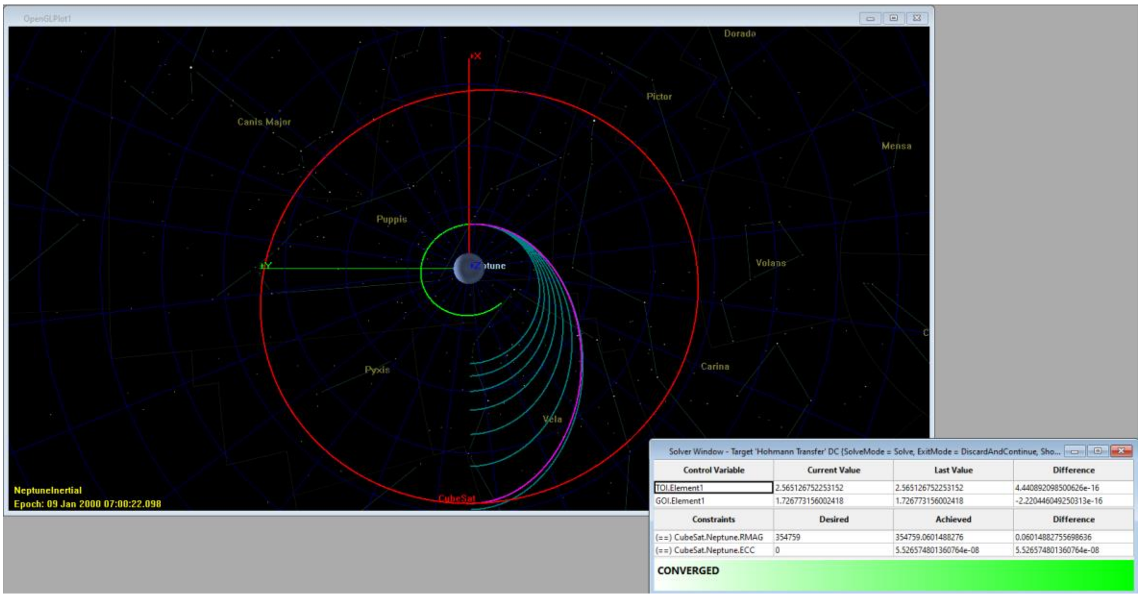



*Figure 43. Mission convergence. (Elaboración propia, 2023)*

In light blue, the different iterations to achieve the solution can be seen, the final trajectory being the one highlighted in purple. The last step of the mission sequence, which was to propagate the CubeSat for one day on that orbit, is shown in red, with Neptune in the middle. The mission can also run an animation to visualize the full steps followed until achieving the targeted and desired orbit.

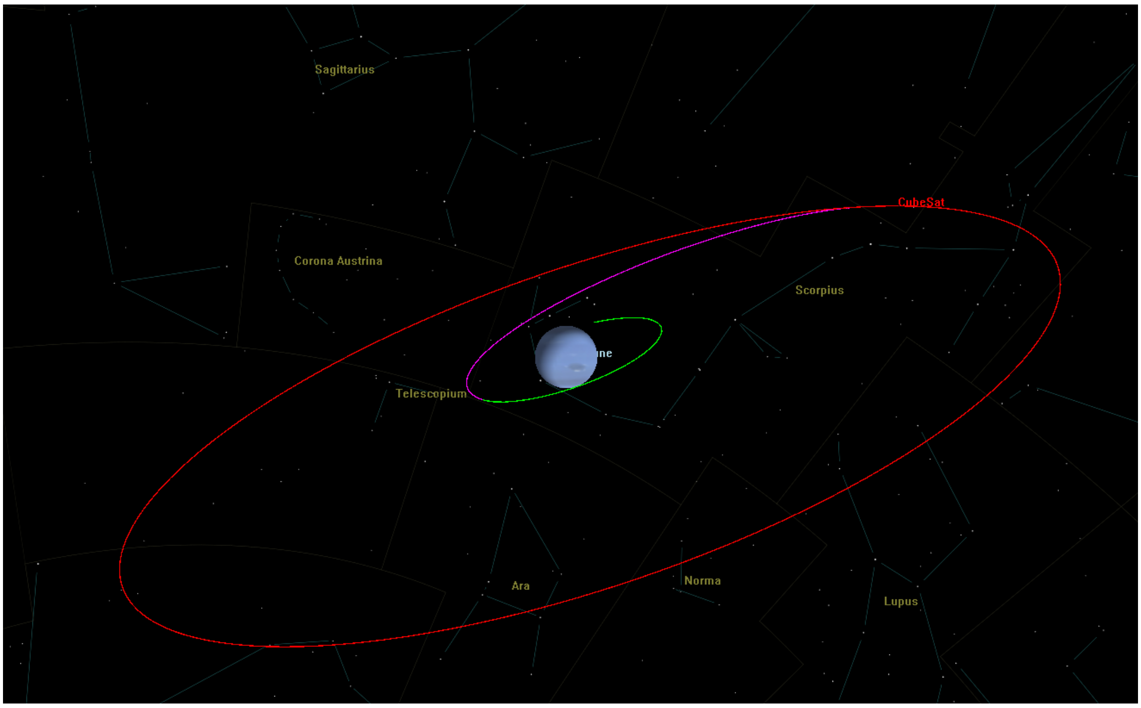



*Figure 44. CubeSat achieved orbit. (Elaboración propia, 2023)*

Finally, in Figure 44, the final orbit achieved by the CubeSat is shown, following that of Triton's, also showing the inclination of that moon with respect to Neptune.

Conclusively, using the GMAT software gives us several possibilities when developing a mission, being able to reproduce a full trajectory, several manoeuvres, orbits, etc, also solving for certain objectives such as inclination or eccentricity. The next step would be reproducing the orbit around Triton itself, as this simulation was to show how the insertion to that moon would take place. The orbit around Triton could not be achieved for this project as there is no Triton model on GMAT, so a plugin may be developed for this purpose in the future.

# Chapter 7. CONCLUSIONS & FUTURE WORKS

During Chapter 1 and Chapter 2, the mission proposed for this research was described in a more or less briefly manner, in order to put the reader in perspective of what was wanted to develop and the characteristics and procedures to be followed.

In Chapter 3, the problems faced when developing a deep space mission as the one proposed at the beginning of this report are explained.

In Chapter 4, the main subsystem that is considered to face one of said issues, and the one to be researched during this project is introduced, giving a few possibilities of plausible solutions that are considered a real possibility nowadays.

During Chapter 5, the proposition of research and investigation for the project at hand is introduced, developed and explained. Its mode of operation, as well as the different missions in which RTGs have been used are discussed. Besides, several previous and existing models are mentioned and compared, trying to focus on one possible model to follow as an example to design the radioisotope thermoelectric generator desired for the CubeSat utilized for the mission proposed, orbit Triton.

According to the research performed and the possibilities given, developing a small RTG applicable to small satellites such as a CubeSat is more than plausible. Not only because its adaptability is viable, but also because the output power provided would facilitate disregarding other power sources such as solar panels and batteries, decreasing the mass constraint and giving more options in other subsystems. Moreover, some studies mentioned in Chapter 8 have started getting deep in this possibility, so the future of the development of small RTG looks bright if proper fund is given to the sector in order to improve and experiment.

Finally, in Chapter 6, a full simulation following the trajectory and orbit transfer to be performed by the CubeSat, being launched from the mothership orbiting Neptune, is shown and explained, with accurate and precise results and excellent visualization of what that trajectory would look like.

As for the future works regarding this whole project, the next step should include the mentioned orbit around Triton, developing a plugin to be able to input a Triton model in GMAT. Continuing in the orbit section, a full trajectory from Earth to Neptune could be simulated to visualize the path followed by the mothership and all the physics it would imply.

Regarding the RTG researched, a possible next step would be fabricating it or something similar in performance to observe the results, maybe using MatLab. Also, the full design of the interior of the CubeSat could be done, as well as adding the RTG developed on the outside.

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